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SPACE STATION WP-04 POWER SYSTEM

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PRELIMINARY ANALYSIS & DESIGN DOCUMENT
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4.0 SPECIFICATION TREE

The hierarchy of specifications defined in Table 4-1 represents the flow down of contract technical specification requirements from the Space Station Program Definition and Requirements Document (JSC 30000), to the specifications prepared in support of the acquisition (design, performance, testing, procurement, and delivery) phase, for the Space Station Electric Power System (EPS).

Rocketdyne plans to generate a system level specification for the EPS in order to facilitate the usage, accountability, and tracking of overall system level requirements.

Five contract end item (CEI) specifications will be generated as shown in Table 4-1. These specifications establish the technical program requirements baseline for the five contract end items which make up the EPS.

Prime Item Development/Procurement specifications will be prepared to specify the requirements for qualification, production, quality control, acceptance verification, and preparation for delivery of assemblies, ORUs, and components, as appropriate.

Manufacturing process specifications and long lead material procurement specifications will be prepared to support manufacturing operations and schedules. Assembly, checkout and startup specifications will be prepared as required to support engineering, test, manufacturing, quality assurance, delivery, and operational requirements. Operation and maintenance manuals will be prepared as required by contract.

An analysis of the ORUs making up each CEI has been performed and is reflected in the Space Station EPS Drawing Tree shown in Figure 4-1 (sheets 1 through 10). Preliminary make or buy information is given for each block and those blocks requiring a specification are identified by an "S" designation near the lower right hand corner. Note that specifications are planned at a variety of hardware levels (i.e., assembly, ORU, component) depending upon the specific make or buy determination of each assembly and its ORUs/components.

Dotted lines within a solid block indicate that the item will be developed within another CEI, but will also be used within the CEI indicated. Dotted lines alone indicate that the item will be developed within the same CEI, but is used in more than one place. In these cases only one specification will be prepared.

Software requirements will be included in the CEI and Prime Item Development/Procurement specifications as applicable. Flight support and ground support equipment items are shown for the Photovoltaic (PV), Solar Dynamic (SD), and PMAD subsystems. Specifications for flight and ground support equipment items will be prepared as required.

TABLE 4-1

Hierarchy Of Specifications
For Space Station EPS

Program Level	Space Station Program Definition and Requirements Document JSC 30000.
System Level (MIL-STD 490A Type A Format)	System Specification for the Space Station Electric Power System.
Contract End Item (CEI) Level MIL-STD 490A Type B Format)	Specifications for the design, performance, environment, interfaces and verification for each of the five contract end items: 1) Station PV Module 2) Station SD Module 3) Station PMAD Subsystem 4) Platform PV Subsystem 5) Platform PMAD Subsystem.
Prime Item Development/Procurement Level (MIL-STD 490A Type B1 Format)	Specifications for the production, qualification, quality control, verification, and preparation for delivery of assemblies, ORUs, and components within each of the five contract end items. (See Figure 4-1).

<u>Software Development And Procurement</u> (MIL-STD 490A, Type B5/C5 Format)	Software development/procurement specifications as appendices to the Prime Item Development/Procurement specifications.
<u>Support Equipment</u> (MIL-STD 490A, Optional Format)	Ground support equipment (GSE) and flight support equipment (FSE) procurement specifications as required to support system and subsystem requirements.
<u>Manufacturing Processes</u> (MIL-STD 490A, Type D Format)	Process specifications as required to define equipment, materials, and manufacturing processing requirements.
<u>Material Procurement</u> (MIL-STD 490A, Type E Format)	Material specifications for the fabrication of make items to specify the functional, physical, chemical, electrical, and mechanical requirements of material being procured.
<u>Assembly, Checkout and Start-Up</u> (Optional Format)	Specifications or procedures to form the basis for assembly, operations checkout and maintenance.
<u>Operation and Maintenance</u> (Format to be determined by Contract)	Operations and maintenance manuals in accordance with contract requirements.

SHEET 5

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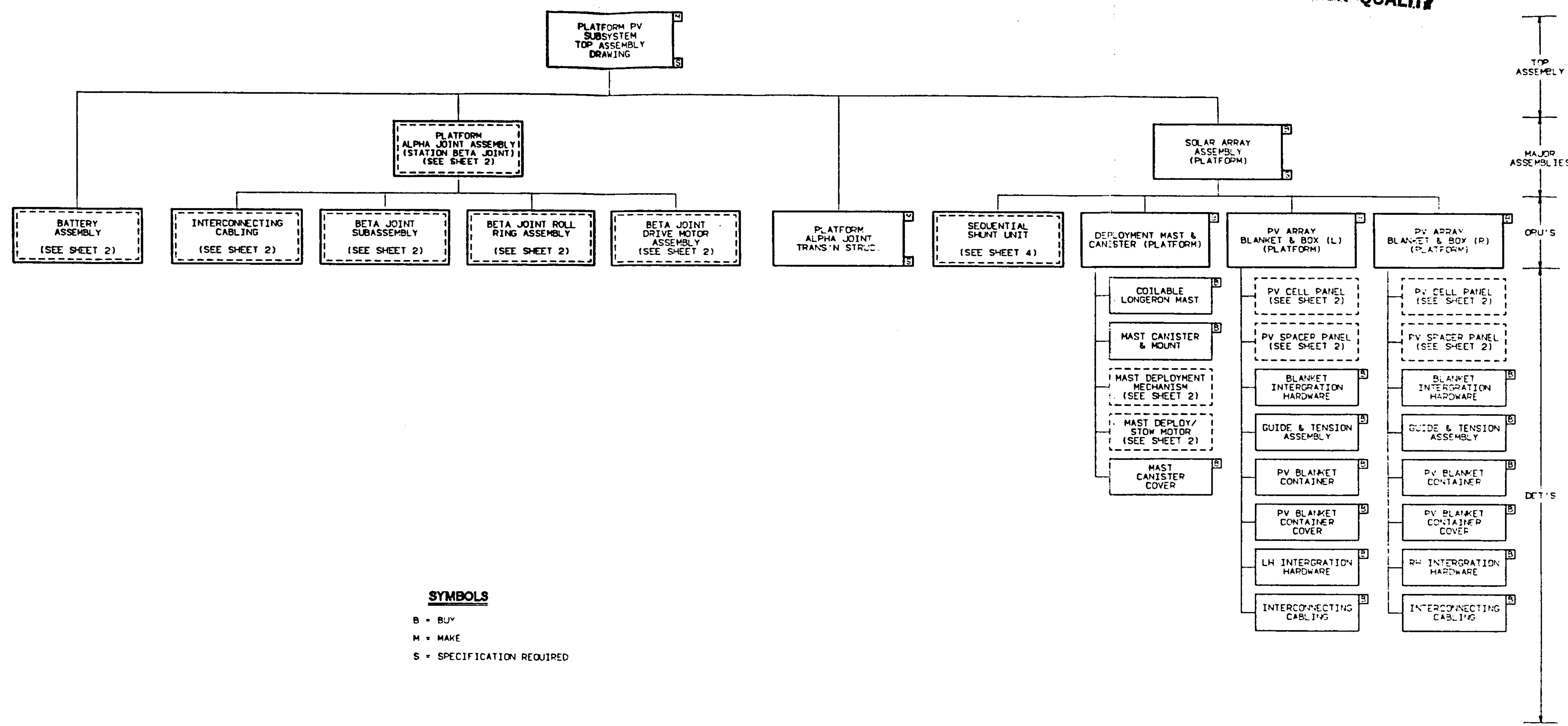


SYMBOLS

B = BUY

M = MAKE

S = SPECIFICATION REQUIRED

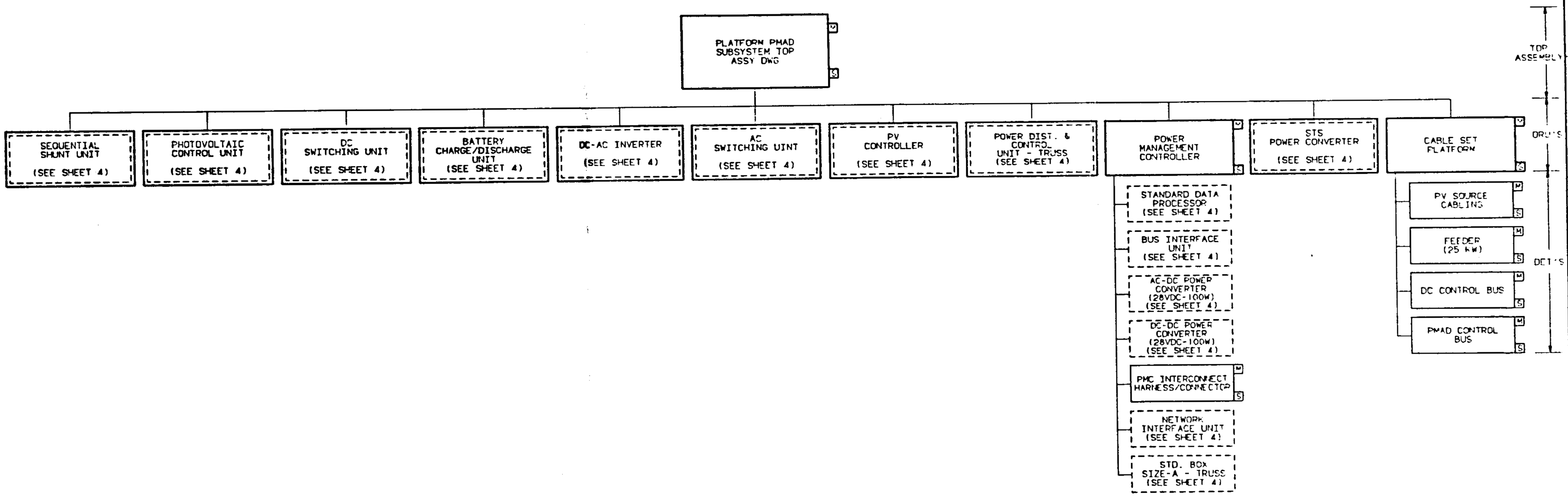
FOLDOUT FRAME

FOLDOUT FRAME

FIGURE 4-1 (CONTINUED)

PLATFORM PV SUBSYSTEM DRAWING TREE

SHEET 6



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SYMBOLS

M = MAKE

S = SPECIFICATION REQUIRED

FOLDOUT FRAME

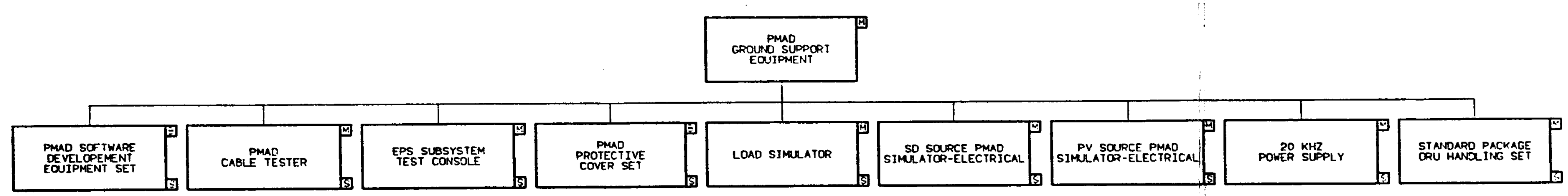
FOLDOUT FRAME

FIGURE 4-1 (CONTINUED)

PLATFORM PMAD SUBSYSTEM DRAWING TREE

E 02602 7R070050

SHEET 10



FOLDOUT FRAME

SYMBOLS

B = BUY
M = MAKE
S = SPECIFICATION REQUIRED

FOLDOUT FRAME

FIGURE 4-1 (CONTINUED)

PMAD GROUND SUPPORT EQUIPMENT

E 02002

5.0 SYSTEMS TEST AND VERIFICATION PLAN
WORK PACKAGE LEVEL INTERFACE APPROACH

5.1 INTRODUCTION

Submittal requirements for the "System Test and Verification Plan", specify that only interfaces between work packages at the subsystem level be included. Therefore, the plan has been structured to focus upon this specific facet of the overall test and verification process. The solar power module physical interfaces are shown in Figure 5-1.

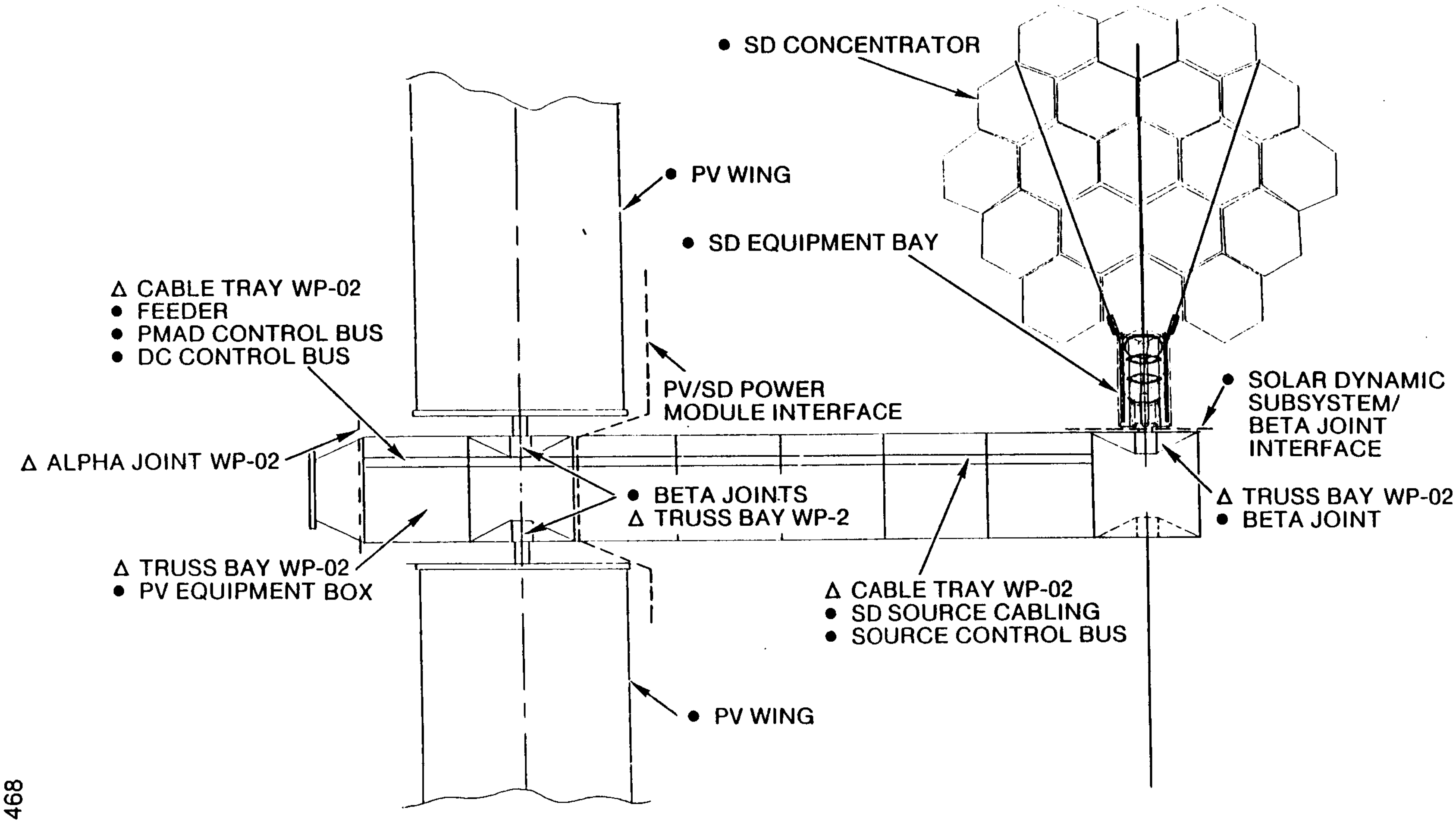
The inclusion of test and verification (T & V) considerations early in the design and development process is an important measure in assuring the ultimate success of the Space Station program. Verification activities planned by Rocketdyne will begin at the individual piece part and component level and progress to subassemblies, assemblies, subsystems, and finally integrated systems.

Section 5.2 traces the origins and status of the Rocketdyne verification planning effort and provides an overview of Space Station program interactions.

Section 5.3 outlines the work package level interfaces between the Electric Power System (EPS) and the other Space Station work packages. Also considered are interfaces with other key program elements including ground support equipment (GSE), the NSTS, and the Space Station crew. Interfaces of both a physical and functional nature, and hardware and software, are discussed.

Section 5.4 contains Rocketdyne's planned approach to verification of physical, functional, and software interfaces. The use of the interface control document (ICD), formalized plans, and master gauges are emphasized in this section as well as the use of process simulators for functional and software interface verification.

SOLAR POWER MODULE EXTERNAL INTERFACES



• WP-04 HARDWARE
Δ PHYSICAL INTERFACE WITH OTHER WP

86D-13-1468

Figure 5-1 Solar Power Module External Interfaces

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Rocketdyne Division

The section also reviews Rocketdyne's preliminary analyses in a number of areas, as they relate to work package level interface verification. Other subjects covered in this section include the use of simulators, ground vs. on-orbit verification, the prototype vs. protoflight approach, growth interface verification, and the use of built-in test equipment (BITE). A process flow diagram is given for the T & V activities on each EPS subsystem.

5.2 VERIFICATION PROGRAM OVERVIEW

The verification of interface design and functional requirements will be the result of synthesizing data from two primary sources; Combined Elements and Integrated Systems (CE&IS) generated requirements and WP-04 generated requirements. All requirements will be evaluated and assigned to one (or more) of the Phase C/D program verification processes as shown in Figure 5-2. Additional analysis will be performed to assign verification levels, program phase relationship, inter-work package support requirements, certification requirements, etc. These will form the basis for writing detailed WP-04 verification plans.

5.2.1 Verification Requirements

A major portion of the external-to-WP-04 system interface verification effort will be performed under the auspices of the CE&IS verification process.

Combined Elements and Integrated Systems verification is the verification of the program requirements from the following sources:

Space Station Program Definition and Requirements (PDRD), JSC-30000, Section 3, Space Station Systems Requirements

Program Level ACDs

Program Level ICDs

Master Measurement List

Other requirements originated at the program level and which are assigned by the Verification Program Office for CE&IS verification.

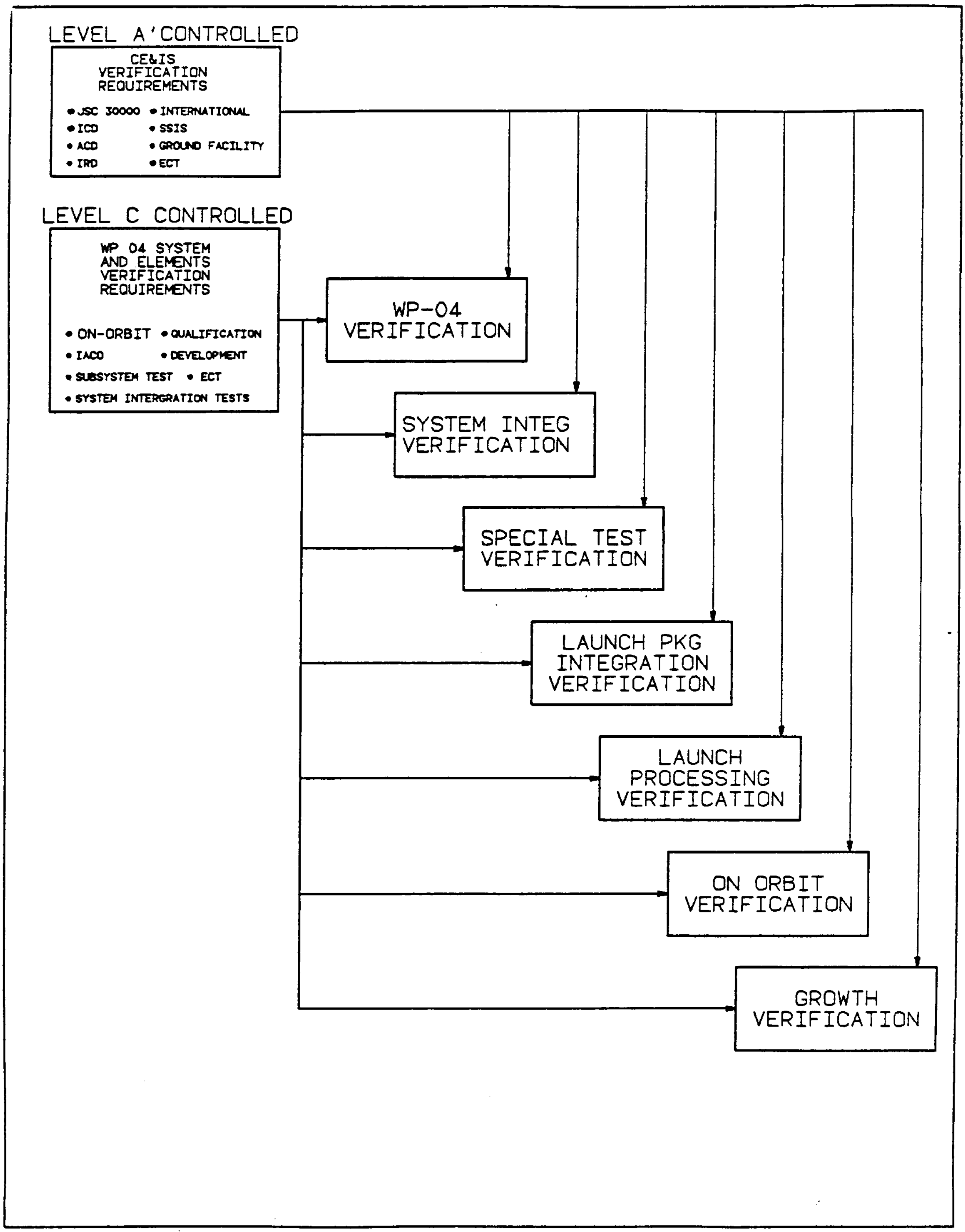

Figure 5-2 Verification Process Flow

The initial CE&IS-MVP addresses only those requirements in the PDRD, Section 3, issue dated April 15, 1986, including changes through Change 3, dated August 15, 1986. As other requirements from sources above are defined, they will be added to the CE&IS-MVP and passed on to the WP-04 MVP.

CE&IS verification will be planned as a cumulative process which will be performed in increments, as shown in Figure 5-2. As can be seen, these increments correspond to the progress of Space Station development beginning with development activities at the WP-04 level and continuing until the performance of a complete system can be evaluated on orbit. The growth increment will accommodate plans for the verification requirements which extend the initial station capability. The increments are defined as follows and bear a time phase relationship to the verification phases discussed henceforth:

WP/SSPP Verification - CE&IS verification tasks usually of an interface of analysis nature which can best be done at the WP/SSPP level, e.g., at the WP/SSPP contractor's plant before shipment of a subsystem.

Systems/Element Integration - CE&IS verification tasks involving systems, subsystems and elements, including hardware and software.

Special Tests - CE&IS tests such as major ground tests, shuttle flight tests, special environmental tests, and other one-time tests which cannot be accommodated by the Systems/Element Integration function above.

Launch Package Integration Verification - CE&IS verification which involves ground integration of the flight hardware and software which make up a particular launch package or combinations of launch packages.

Launch Processing Verification - CE&IS Verification at the launch site.

On-orbit Verification - CE&IS verification includes assembly and checkout on-orbit and the evaluation of Systems/Elements during operations.

Growth Verification - CE&IS verification of the Space Station as additions, deletions, and modifications are made during the growth phase.

CE&IS verification is planned such that a program requirement will be verified in more than one of the above increments. Each CE&IS verification task in each increment will be assigned to a single organization as "prime" responsibility. The prime assignee, usually a WP or SSPP, will be responsible for performing the verification task and acknowledging completion to the CE&IS Verification Program Office (Level A').

Figure 5-3 depicts the current hierarchy of the Space Station program verification documents. Review of the Level A' CE&IS "Verification Requirements", and "Verification Implementation Requirements" documents was completed. It is recognized that the document will be addended and updated by change request to the SSCB, and the WP-04 MVP will reflect those approved allocations as applicable.

Initial CE&IS system level verification requirements for the EPS derived from the PDRD, Section 3, para. 2.2.3, "Electrical Power Systems" requirements are listed in Table 5-1. This has been baselined in the PDRD Section 2. Other WP-04 CE&IS verification responsibilities are defined in JSC-30000 as part of other systems (e.g., TCS, GN&C, DMS, FMS, etc.). These responsibilities will also result in specific WP-04 activities.

Detailed planning of these basics will be integrated with WP-04- originated requirements as described above. The CE&IS activity is not independent of the WP-04 verification task.

Although CE&IS process originates with "top level" (e.g., PDRD, program level ACDs and ICDSs, etc.) requirements, it is currently planed to carry all verification requirements in the WP-04 VDB, thus permitting the efficient satisfaction of common requirements generated from within the WP-04 and externally, from the CE&IS program.

Activities involving WP-04 interface verification conducted at other WP/SSPP locations will be selectively monitored by Rocketdyne.

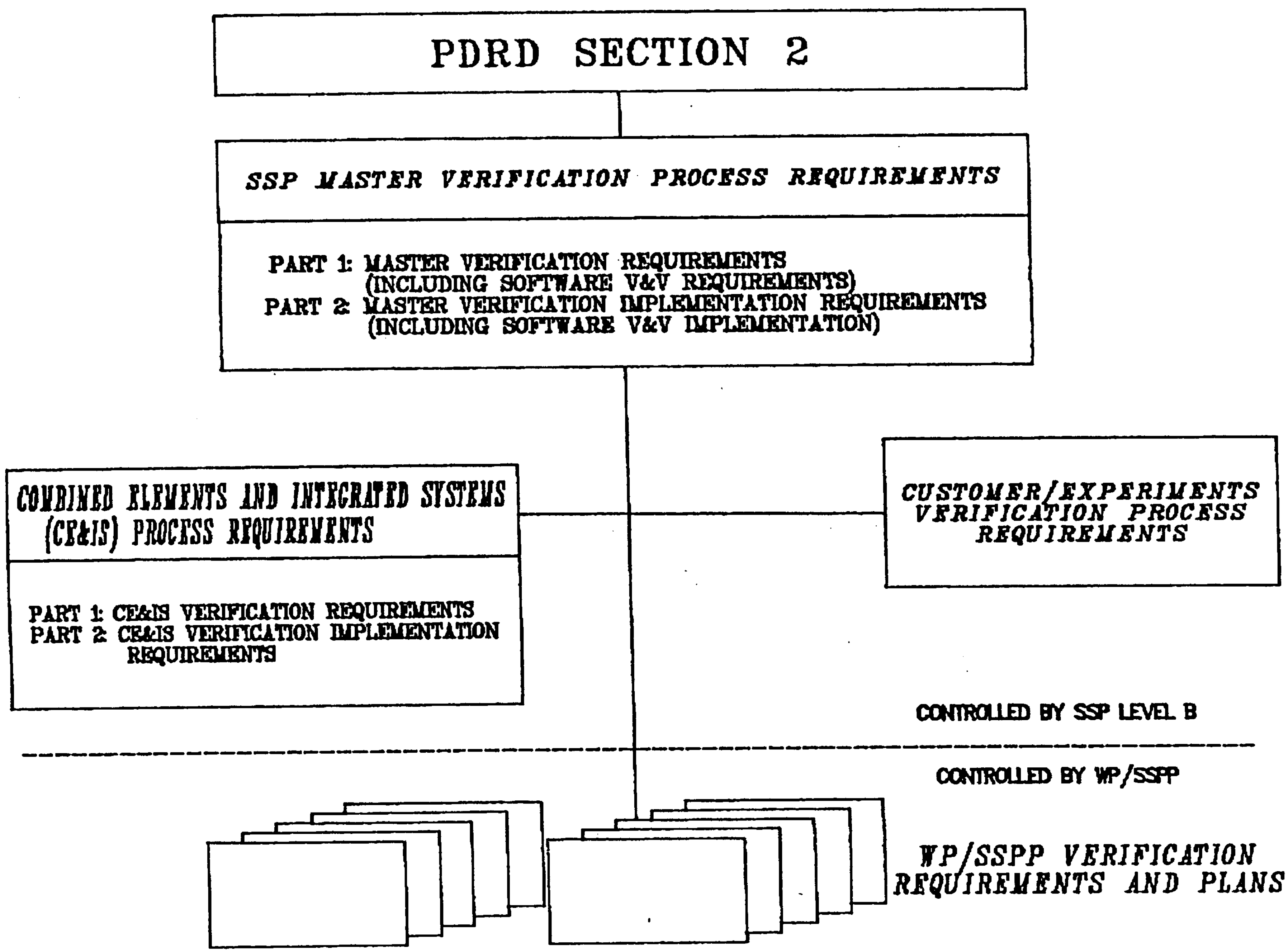

Figure 5-3 Hierarchy of Program Verificaiton Documents

TABLE 5.1. SYSTEM LEVEL VERIFICATION REQUIREMENTS

VERIFICATION REQUIREMENT
2.2.3. ELECTRICAL POWER SYSTEM (EPS) The functions of the EPS consist of power generation energy storage and Power Management and Distribution (PMAD). The EPS system characteristics are shown in table 2-22*, EPS Characteristics.
2.2.3.1. FUNCTIONAL REQUIREMENTS 2.2.3.1.1. SPACE STATION Verify that the EPS provides all IOC user and housekeeping electrical power and provides for the capability to incrementally increase the power available to meet the increasing power needs for growth of the Space Station.
2.2.3.1.2. PLATFORM Verify that the EPS provides all IOC user and housekeeping electrical power for the platforms and shall have the capability to incrementally increase the power available to meet growth power requirements.
2.2.3.1.3. SAFE HAVEN Verify that the EPS provides power to meet safe-haven requirements. Verify that the EPS meets the safe-haven requirements with built-in redundancy.
2.2.3.1.5. CONTINGENCY POWER Verify that the EPS is capable of providing power for one orbit (with no solar input) such that the Station may be returned to normal operation at the end of that period. This orbit shall start at the beginning of a normal sunlight period. A power down to minimum safe levels may be utilized to minimize the impact of this requirement on the EPS. This energy requirement shall be a minimum of 15.0 kW hours. For the growth Station, this requirement shall be maintained at the IOC level. Table 2-18* details the Space Station power capability.
2.2.3.1.6. NATIONAL SPACE TRANSPORTATION SYSTEM (NSTS) POWER TRANSFER Verify that the EPS has the capability to transfer power to or receive power from the NSTS orbiter.
2.2.3.1.7. POWER SYSTEM CONTROL Verify that a capability is provided which shall monitor, evaluate, and control the EPS performance from sources to the interface to the loads; detect, isolate, and clear faults; provide alerts for emergencies; and control loads by an appropriate load-shedding technique. The capability shall maximize the utilization of the EPS. Verify that the crew is able to override automatic and autonomous controls consistent with safety requirements.

*See PDRD section 3.

TABLE 5.1- SYSTEM LEVEL VERIFICATION REQUIREMENTS

VERIFICATION REQUIREMENT
2.2.3.1.8. POWER PROTECTION 2.2.3.1.8.A. Verify that power protection is provided for the power sources against overloads or faults in distribution and user subsystems. 2.2.3.1.8.B. Verify that power protection is provided for the distribution function against overloads and faults in the power system. 2.2.3.1.8.C. For the user loads, verify that power protection is provided to prevent adverse impact or damage to other user subsystems.
2.2.3.1.9. MAIN BUS INTERCONNECT CAPABILITY 2.2.3.1.9.A. Verify that means of selecting, connecting and disconnecting the sources of electrical energy to the vehicle main electrical buses are functional. 2.2.3.1.9.B. Verify that means of selecting, connecting, and disconnecting between main electrical buses and loads are functional.
2.2.3.2. DESIGN AND PERFORMANCE REQUIREMENTS 2.2.3.2.1. SPACE STATION POWER LEVEL 2.2.3.2.1.A. Verify that at IOC the EPS has the capability to continuously provide 50 kilowatts (kW) for the users and 25 kW for housekeeping. 2.2.3.2.1.B. Verify that the growth EPS has the capability to continuously provide 250 kW for the users and 50 kW for housekeeping. 2.2.3.2.1.C. The Station power capability is detailed in table 2-18* for the Main-Tended Approach (MTA), IOC, and growth stations.
2.2.3.2.2. PLATFORM POWER LEVEL The platform power capability is detailed in table 2-24* for the IOC and growth configurations.
2.2.3.2.3. EPS "USER" INTERFACE 2.2.3.2.3.A. Verify that the EPS provides 20 kHz, 208 volts, single phase sine wave, utility grade power to the user interface for use in the user's equipment. 2.2.3.2.3.B. This standard ac power shall be used by all U.S. and international partner elements of the SSP including manned base modules, attached payloads and equipment.

*See PDRD section 3.

TABLE 5.1- SYSTEM LEVEL VERIFICATION REQUIREMENTS

VERIFICATION REQUIREMENT
2.2.3.2.3. EPS "USER" INTERFACE - CONTINUED 2.2.3.2.3.C. Verify that the platform EPS distributes 20 kHz, 1 phase sine wave, 208 volts, utility grade power to the user interface for use in user's equipment. 2.2.3.2.3.D. Verify that the rated system capability is the sum of all the power delivered to these user interfaces.
2.2.3.2.4. EPS BUS VOLTAGE 2.2.3.2.4.A. Verify that the Space Station EPS uses 20 kHz, 440 volts, single phase sine wave electrical buses for distribution of power. 2.2.3.2.4.B. Verify that the platform EPS uses 20 kHz, 440 volts, single phase sine wave electrical power for distribution.
2.2.3.2.5. GROUNDING Verify that the power system provides a single point ground. That the grounding schemes used on all SSPEs is compatible.
2.2.3.2.6. POWER MANAGEMENT AND DISTRIBUTION (PMAD) REDUNDANCY AND FAULT DETECTION 2.2.3.2.6.A. Verify that diverse routing of redundant wiring and location of redundant components has been implemented. 2.2.3.2.6.B. Verify that power is fed through the wall of each habitable element in at least two physically separate locations. Each feedthrough shall be capable of providing the full module power. 2.2.3.2.6.C. Verify that all wiring is short circuit protected with replaceable or resettable devices or be current limited.
2.2.3.2.7. POWER SYSTEM POINTING REQUIREMENTS Verify that the Space Station has rotating joints capable of holding the power generation devices in a solar inertial orientation and transferring electrical power and power system data for the assembly, IOC, and growth phases. See table 2-19*, Power Generation Subsystem (PGS) Pointing Requirements. The data in table 2-19 are composite requirements on the alpha/beta/fine pointing combination of joints.

*See PDRD section 3.

TABLE 5.1- SYSTEM LEVEL VERIFICATION REQUIREMENTS

VERIFICATION REQUIREMENT
2.2.3.2.7.1. ALPHA AXIS ROTARY JOINT EPS REQUIREMENTS Verify that the Space Station has rotating joints that are, with 360 degree continuous rotation, capable of making the PGS alpha axis correction and transfer of electrical power and power system data for the assembly, IOC, and growth phases. See table 2-20*, Alpha Axis Rotary Joint EPS Requirements.
2.2.3.2.7.2. BETA AXIS ROTARY JOINT EPS REQUIREMENTS Verify that the Space Station has rotating joints capable of making the beta axis correction and transferring electrical power and power system data during assembly, IOC, and growth phases. See table 2-21*, Beta Joint Electrical and Data Requirements. Verify that Beta joints are compatible with Solar Dynamic (SD) and Photovoltaic (PV). Power and data transfer requirements are stated on a per side (wing) basis.
2.2.3.2.8. EPS GROWTH Verify that the EPS design provides growth capability and onorbit reconfiguration capability to accommodate growth in power demand on the Station and changes in the distribution of power needs at various places on the Station. (See table 2-18*) The EPS distribution system shall be sized and installed for 175 kW at IOC.
2.2.3.3. EPS RELIABILITY REQUIREMENTS The EPS shall satisfy the following redundancy/reliability requirements: 2.2.3.3.6. Verify that two redundant power penetrations are provided through the wall of each habitable module (including logistics modules) at physically separate locations to maximize reliability. Each shall be capable of providing 25 kW to the module.
2.2.3.4. ELECTRICAL POWER SYSTEM (EPS) THERMAL CONTROL REQUIREMENTS 2.2.3.4.1. Verify that thermal control (outboard of alpha joint) for the Solar Power Module (SPM) is accomplished by local, autonomous thermal control subsystems. The SPM EPS systems/components shall include but not be limited to the following: 2.2.3.4.1.A. PV power generating system 2.2.3.4.1.B. SD power generating system 2.2.3.4.1.C. NiH2 batteries

*See PDRD section 3.

TABLE 5.1- SYSTEM LEVEL VERIFICATION REQUIREMENTS

VERIFICATION REQUIREMENT
2.2.3.4.1. CONTINUED 2.2.3.4.1.D. Power conditioning and control 2.2.3.4.1.E. Parasitic loads
2.2.3.4.2. Verify that thermal control (inboard of alpha joint) for the power components is accomplished by utilizing the Station central TCS. The power components shall include, but not be limited to, the following: 2.2.3.4.2.A. Bus switching assembly 2.2.3.4.2.B. Power distribution control assemblies 2.2.3.4.2.C. Transformers
2.2.3.4.3. Verify that thermal control for the platform EPS is achieved by utilizing the platform TCS. The platform EPS shall include but not be limited to the following: 2.2.3.4.3.A. PV power generating system 2.2.3.4.3.B. NiH2 batteries 2.2.3.4.3.C. Power conditioning, control, and distribution

5.2.2 Verification Data Base

A verification data base (VDB) will be established to document development, qual and system test requirements for every level of hardware on each ORU/subassembly. The matrix will be programmed in R-Base System 5 and will permit access to such information as:

- Source reference (requirement document)
- Plan/DWG/SPEC Number
- Objective (specific) may be more than one
- Verification type ($\geq$ 1)

Devel	Systems
Qual	Interface
Certification	CE&IS
Acceptance	IACO

- Hardware type (brassboard, prototype, etc.)
- Quantity
- Hardware need dates and location
- Activity date and location
- Category (analysis, inspection, demo, test)
- General test method/medium
- Estimated test duration
- Retest requirement
- Partial or complete test
- Certification requirement information
- Support needs/supplier/type
    - Hardware
    - Software
    - TSE
    - FSE
    - GSE
    - GFE

Scheduling of external interface verification activities will be visible to the extent required to allow other WP/SSPP participation as approved by Level A'.

The VBD is an extension of the Rocketdyne EPS data base and allows commonality/compatibility with other WP/SSPPs and the SSP verification office. Although all verification requirements will be controlled by the WP-04 Master Verification Plan (MVP, LeRC # TBD), the information contained in the VDB will be retrievable to support SSP major reviews (PDRs, CDRs, DCRs, Acceptance, ORRs, FRRs, etc.). The VDB will also allow the periodic statusing of the EPS verification effort and scheduled activities.

Table 5-2 contains details of that EPS hardware currently identified in the ICD as having on-orbit operating physical/functional interfaces with external WPs. It contains data which will form the basis for planning specific verification activities and identifying and support equipment/hardware requirements, including that needed from other WPs. Similar data will be generated for all interfaces; checkout, IACO, CE&IS, NSTS, launch, etc.

It is Rocketdyne's intention to confirm the adequacy of all interface design/functional requirements as early as it is feasible in the manufacturing and test flow. Some items will be incrementally verified and complete certification may be a sum of several individual tasks. For example, electrical (AC and DC) interfaces of the MBSU will be verified at the component development level using WP-02 specification requirements, while final verification of EMI compatibility may be deferred to on-orbit. The remainder of the indicated elements would be verified at subsystem/system level activities.

At this time, the physical configuration of the platforms (POP and COP) are not sufficiently defined to identify specific WP-04 hardware placements. The data will be incorporated into the EPS data base as soon as it is available, and specific interface verification plans generated for the PV and PMAD hardware located on the platforms.

Verification tasks are planned by five methods; similarity, analysis, demonstration, inspection, and test. These methods have been defined by the program for application by all verification functions and are as follows:

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Space Station . PMAD Subsystem . Main Bus Switching Unit		WP02-Utility Plate-Inboard Truss								
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage		i	d	-	-	-	-		
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5- 2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Space Station . PMAD Subsystem . PDCU-Truss		WP02-Utility Plate-Inboard Truss	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage		i	d	-	-	-	-		
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
s = similarity
a = analysis
i = inspection
d = demonstration
t = test
- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Space Station . PMAD Subsystem . PDCU-Module/node		WP02 Elec Rack-Pressurized Module	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage		i	d	-	-	-	-		
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC								X	
	Intra-Vehicular Act		t	t	-	-	-	-		
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Space Station . PMAD Subsystem . Transformer		WP02 Utility Plate Inboard Truss	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection								X	
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage		i	d	-	-	-	-		
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect								X	
	Data/Signal								X	

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Space Station . PMAD Subsystem . Power Management Controller		WP02 - Utility Plate-Inboard Truss								
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage		i	d	-	-	-	-		
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Space Station . PMAD Subsystem . NSTS Power Converter		WP02 - Utility Plate-Inboard Truss								
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage		i	d	-	-	-	-		
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5- 2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Space Station . PMAD Subsystem . Cable Set		WP02 Cable Trays, Utility Plates, Notes,& Electric Racks Inboard	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection								X	
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic								X	
	EMI		t	t	-	-	-	-		
	EVA		t	t	-	-	-	-		
	FTS/MSC								X	
	Intra-Vehicular Act		t	t	-	-	-	-		
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Station . PV Module . Beta Joint Transition Structure		WP02 Outboard Truss	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		s	t*	i	i	d	-		*Testing parameter derived from SD/beta joint interface loads
	Thermal Connection								X	
Envelope	Clearance for Maintenance		s	t*	-	-	d	-		
	Spare Stowage								X	
Environmental	Dynamic		a	t*	-	-	-	-		
	EMI								X	
	EVA		s	s	-	-	-	-		
	FTS/MSC								X	
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect								X	
	DC Control Connect								X	
	Data/Signal								X	

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5- Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Station . Beta Joint . Transition Struct to WP02 Outboard Truss		WP02 Outboard Truss	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		s	t*	i	i	d	-		*Testing parameter derived from SD/beta joint interface loads
	Thermal Connection								X	
Envelope	Clearance for Maintenance		s	t*	-	-	d	-		
	Spare Stowage								X	
Environmental	Dynamic		a	t*	-	-	-	-		
	EMI								X	
	EVA		s	s	-	-	-	-		
	FTS/MSC								X	
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect								X	
	DC Control Connect								X	
	Data/Signal								X	

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Station . PV Module . PV Equipment Box Structure Set		WP02 Outboard Truss	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		a	d	d	d	d			
	Thermal Connection		a	d	d	d	d			
Envelope	Clearance for Maintenance		a	a	-	d	-	-		
	Spare Stowage								X	
Environmental	Dynamic		t	t	-	-	-	-		
	EMI								X	
	EVA								X	
	FTS/MSC								X	
	Intra-Vehicular Act		a	a	a	d	d	d		
Elect/ Data	20kHz Power Connect		s	s	s	d	d	d		
	DC Control Connect		s	s	s	d	d	d		
	Data/Signal		s	s	s	d	d	d		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Station . PV Module . Cable Set		WP02 Trays & Conduits	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		d	d	d	d	d	-		
	Thermal Connection								X	
Envelope	Clearance for Maintenance		i	i	i	d	-	-		
	Spare Stowage								X	
Environmental	Dynamic								X	
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC								X	
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	t	t	d		
	DC Control Connect		t	t	t	t	t	d		
	Data/Signal		t	t	t	t	t	d		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Platform . PMAD Subsystem . Batt Chg/Dschg Unit		WP03-Utility Plates								
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Platform . PMAD Subsystem . DC/AC Inverter		WP03-Utility Plates	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Main-tenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environ-mental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Platform . PMAD Subsystem . PV Controller		WP03-Utility Plates	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Platform . PMAD Subsystem . PVCU		WP03-Utility Plates								
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Platform . PMAD Subsystem . DCSU		WP03-Utility Plates	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5- 2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Platform . PMAD Subsystem .ACSU		WP03-Utility Plates	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Platform . PMAD Subsystem . PDCU-Truss		WP03-Utility Plates	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Platform . PMAD Subsystem .PMC-Truss		WP03-Utility Plates	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Platform . PMAD Subsystem . NSTS Power Converter		WP03-Utility Plates								
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection		t	t	-	-	-	-		
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		t	t	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Platform . PMAD Subsystem . Cable Set		WP03 Cable Trays Utility Plates	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		i	t	-	-	-	-		
	Thermal Connection								X	
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic								X	
	EMI		t	t	-	-	-	-		
	EVA		t	t	-	-	-	-		
	FTS/MSC								X	
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	-	t	t		
	DC Control Connect		t	t	t	-	t	t		
	Data/Signal		t	t	t	-	t	t		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5- 2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Space Station . SD Module . Beta Joint Transition Structure		WP02 Outboard Truss								
Mech/ Struct	Structural Connection*		s	t	i	i	d	-		*Identical to PV beta joint-inter project support hardware required: Truss Node (WP02) EVA IF specs (WP02)
	Thermal Connection								X	
Envelope	Clearance for Maintenance*		s	t	-	-	d	-		
	Spare Stowage								X	
Environmental	Dynamic		a	t	-	-	-	-		
	EMI								X	
	EVA*		s	s	-	-	-	-		
	FTS/MSC								X	
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect								X	
	DC Control Connect								X	
	Data/Signal								X	

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Platform . PV Subsystem . Battery Assy		WP03-Utility Plate								
Mech/ Struct	Structural Connection		i	t	-	d	-	-		
	Thermal Connection								X	
Envelope	Clearance for Maintenance		i	d	-	-	-	-		
	Spare Stowage		i	d	-	-	-	-		
Environmental	Dynamic		t	t	-	-	-	-		
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC								X	
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect								X	
	DC Control Connect								X	
	Data/Signal								X	

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Polar Platform . Alpha Joint Transition Structure		Polar Platform Structure								
Mech/ Struct	Structural Connection		s	t*	i	i	d	-		* Testing Parameters are determined from SD/Beta Joint SS Interface Loads (assumes commonality)
	Thermal Connection								X	
Envelope	Clearance for Main- tenance		s	t*	-	-	d	-		
	Spare Stowage								X	
Environ- mental	Dynamic		a	t*	-	-	-	-		
	EMI								X	
	EVA								X	
	FTS/MSC								X	
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect								X	
	DC Control Connect								X	
	Data/Signal								X	

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
. Co-Orbit Platform . Alpha Joint Trans Struct		Co-orbit Platform Structure								
Mech/ Struct	Structural Connection		s	t*	i	i	d	-		* Testing parameters are determined from SD/Beta Joint SS Interface Loads (assumes commonality)
	Thermal Connection								X	
Envelope	Clearance for Maintenance		s	t*	-	-	d	-		
	Spare Stowage								X	
Environmental	Dynamic		a	t*	-	-	-	-		
	EMI								X	
	EVA								X	
	FTS/MSC								X	
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect								X	
	DC Control Connect								X	
	Data/Signal								X	

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

C-2

Table 5-2 Verification of On-Orbit Operations WP04 External Interfaces

WP04 ITEM		IF WP - HARDWARE	VERIFICATION PHASE							
. Space Station . SD Module . Cable Set		WP02 Cable Trays & Conduits - Outboard	Development	Qualification	Acceptance	Integrated Systems	Pre-Launch	On-Orbit	N/A	Comments
Mech/ Struct	Structural Connection		d	d	d	d	d	-		
	Thermal Connection		-	-	-	-	-	-		
Envelope	Clearance for Maintenance		i	i	i	-	-	-		
	Spare Stowage								X	
Environmental	Dynamic								X	
	EMI		t	t	-	-	-	-		
	EVA								X	
	FTS/MSC		-	-	-	-	-	-		
	Intra-Vehicular Act								X	
Elect/ Data	20kHz Power Connect		t	t	t	t	t	d		
	DC Control Connect		t	t	t	t	t	d		
	Data/Signal		t	t	t	t	t	d		

Legend*
- s = similarity
- a = analysis
- i = inspection
- d = demonstration
- t = test
- \- = not applicable (N/A)

(a) Similarity - A requirement may be verified by similarity if the same "part" has been qualified for a similar application. If all of the requirements have not been verified in this prior verification, it shall only be necessary to verify that these additional requirements are met.

(b) Analysis - Verification by analysis shall be the process of utilizing analytical techniques to verify that the requirements are satisfied. Verification through analysis shall be used when verification by test is not possible, when test introduces significant risk into the system, or when analysis is an appropriate, cost-effective method of verification. In order for a requirement to qualify for verification by analysis, all of the following criteria shall be satisfied:

(1) Verification by inspection is inadequate,
(2) Verification by similarity is inapplicable,
(3) Verification by test carries high risk of damage/contamination of flight hardware,
(4) Analysis techniques that are rigorous and well understood are available, and
(5) Verification by test is not feasible and/or cost effective.

(c) Inspection - Inspection shall be an element of verification consisting of investigation, without he use of special laboratory requirements and shall be generally non-destructive and shall include (but is not limited to) visual inspection, simple physical manipulation, gauging, and measurement.

(d) Demonstration - Demonstration shall be an element of verification that is limited to readily observable functional operation to determine compliance with requirements. This element of inspection does not require the use of special equipment or sophisticated instrumentation.

(e) Test - Test shall be an element of verification that employs technical means including (but not limited to) the evaluation of functional characteristics by use of special equipment or instrumentation, simulation techniques, and the application of established principles and procedures to determine compliance with requirements. The analysis of data derived from test shall be an integral part of this element.

Certification

A significant portion of the program verification activity is certification process, which consists of a series of activities which demonstrate that the equipment being certified will perform within specification limits under specified mission environmental conditions. Certification will generally be required on all ORUs and many components. Piece parts will generally be developed and qualified for the Approved Parts List without formal certification; entire assemblies and subsystems (e.g. PMAD, PV) will normally be qualified without being subjected to complete environmental control, therefore analysis in conjunction with sub-tier certification will lead to verification. Interface requirements will form a part of the certification requirements and the satisfaction of those by test/analysis will be requested before a part is promoted to the certified hardware/software list.

5.3 WORK PACKAGE LEVEL INTERFACES

This section describes the physical and functional interfaces between the Electric Power System (WP-04) and the other Space Station work packages. Figure 5-4 depicts EPS components and their station location. It is recognized that external WP responsibilities are being clarified and the WP-04 external interfaces may change accordingly.

5.3.1 WP-01 Interfaces

Major items included within WP-01 are the Space Station common module, laboratory and logistics modules, environmental control and life support system (ECLSS), and OMV accommodation.

Physical Interfaces

Physical interfaces with the OMV involve the grapple attach points and provisions for replacement of ORUs.

Functional Interfaces

No functional interfaces are planned with WP-01 at the present time.

5.3.2 WP-02 INTERFACES

Major items included within WP-02 are the Space Station assembly structure; manipulators; external thermal control system; EVA systems; guidance, navigation and control system; communication and tracking system; DMS; module outfitting and propulsion elements.

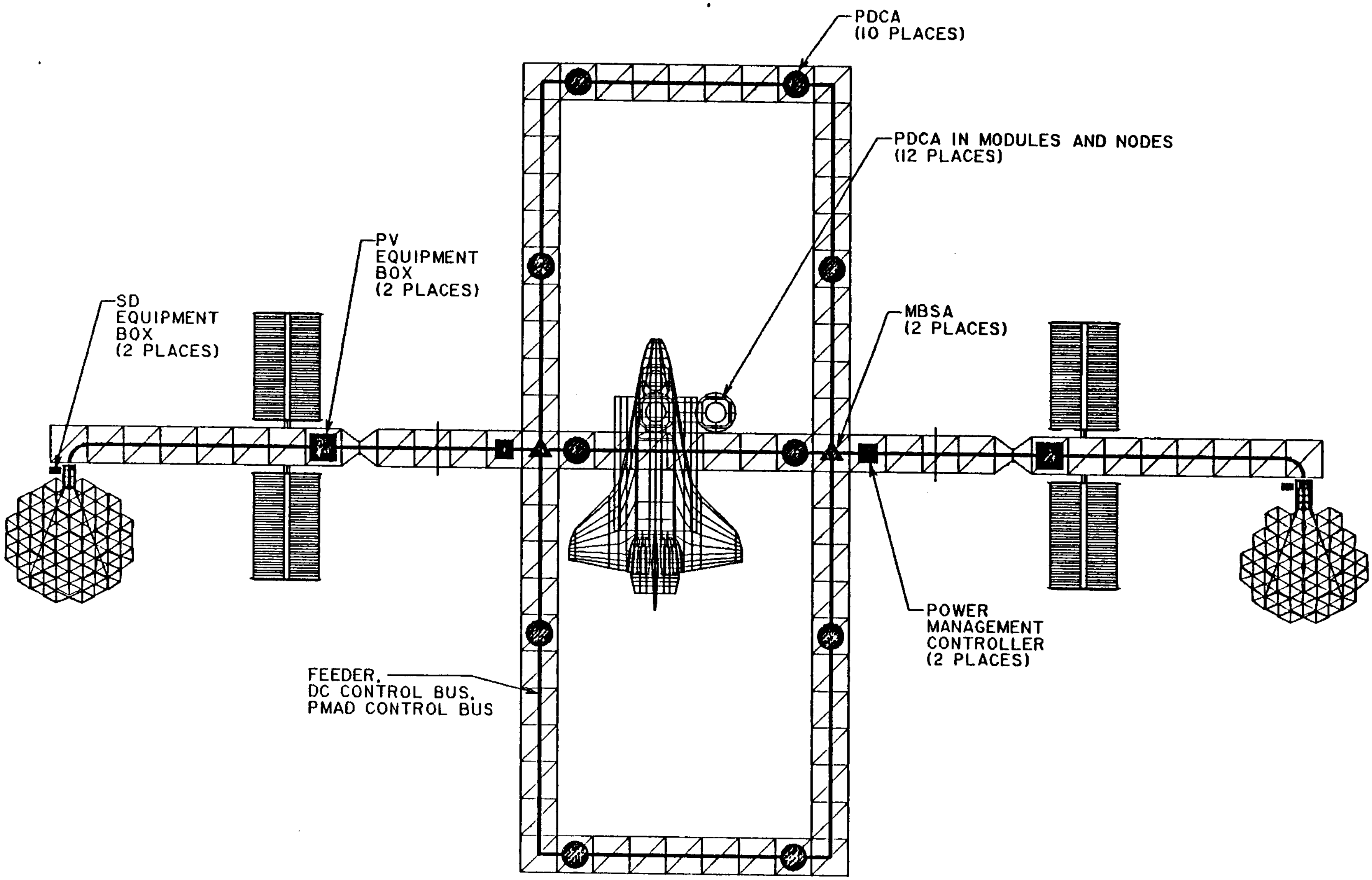

Figure 5-4 EPS Components Locations and Interfaces

Physical Interfaces

Physical interfaces with WP-02 include the following:

- Beta joint with truss structure (Figure 5-5 shows a typical drawing of the beta joint)
- Equipment bay outfitting with the PV equipment box
- PMAD cabling with truss structure
- Grapple attach points and provisions for ORU replacement using manipulators (Figure 5-6 shows a typical PMAD ORU with interfaces)
- Cold plate cooling of PDCAs
- PDCA locations and mounting
- Interface with OMS at power management controllers

Functional Interfaces

Functional interfaces with WP-02 include PMAD. Load characteristics requiring verification include electrical characteristics, duty cycle, automatic control requirements, load shedding priority classification, and operational and casualty mode requirements.

Electrical characteristics include voltage, current, and power factor for both transient and steady state modes. This information is necessary so that the EPS responses to various load conditions can be predicted. Similarly, the characteristics of the EPS must be available to power users so that equipment can be operationally tested.

Duty cycle information includes the expected on and off times and the duration of each. This information will be used for power demand planning and scheduling.

Automatic control requirements may be included in the duty cycle information but if additional modes of operation or if special control methods are required, then these requirements must be integrated and verified.

Load shedding priority classification is required. General categories include crew critical, station critical, payload critical, deferrable, and non-essential.

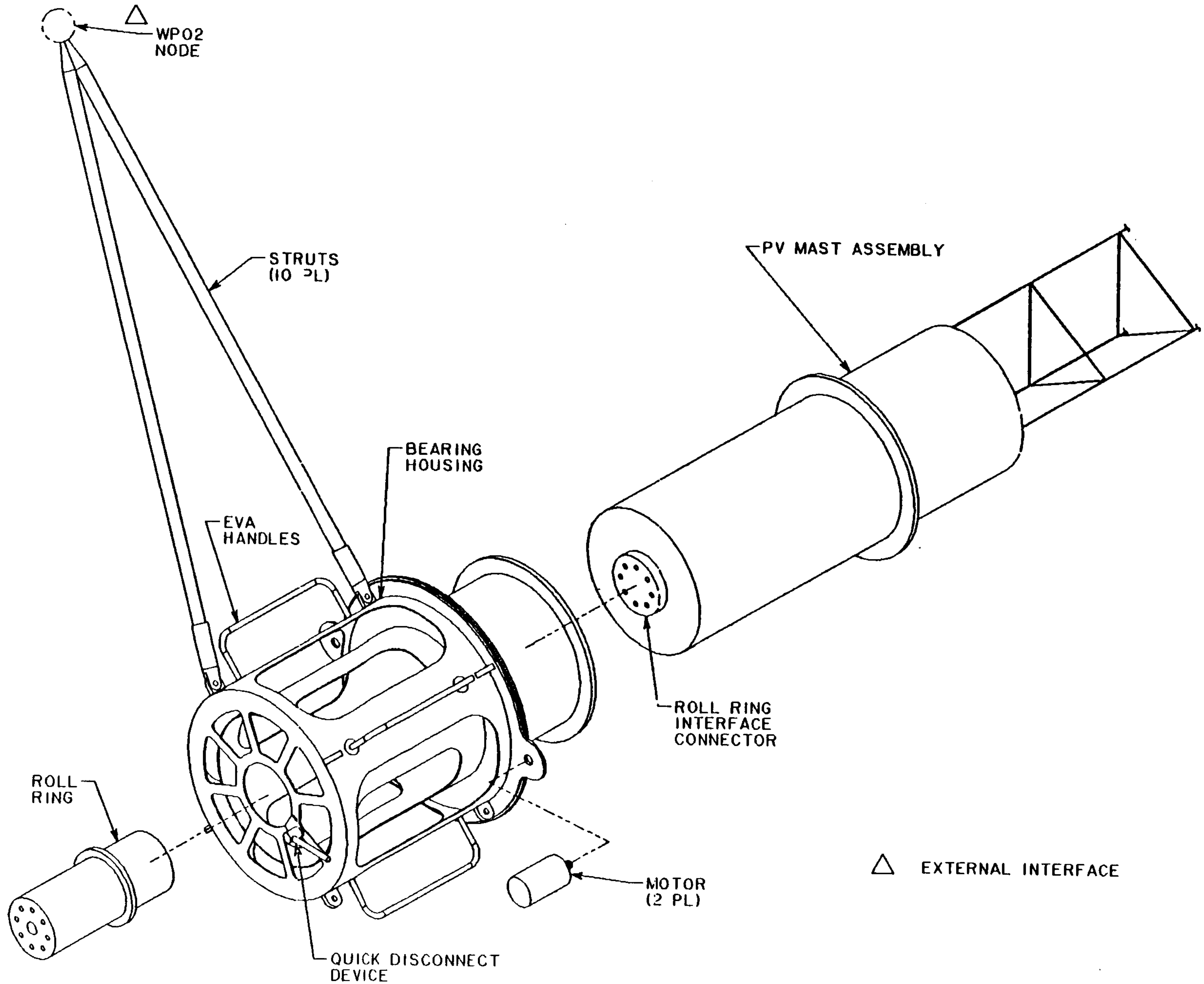

Figure 5-5. Beta Joint, Roll Ring Exploded View

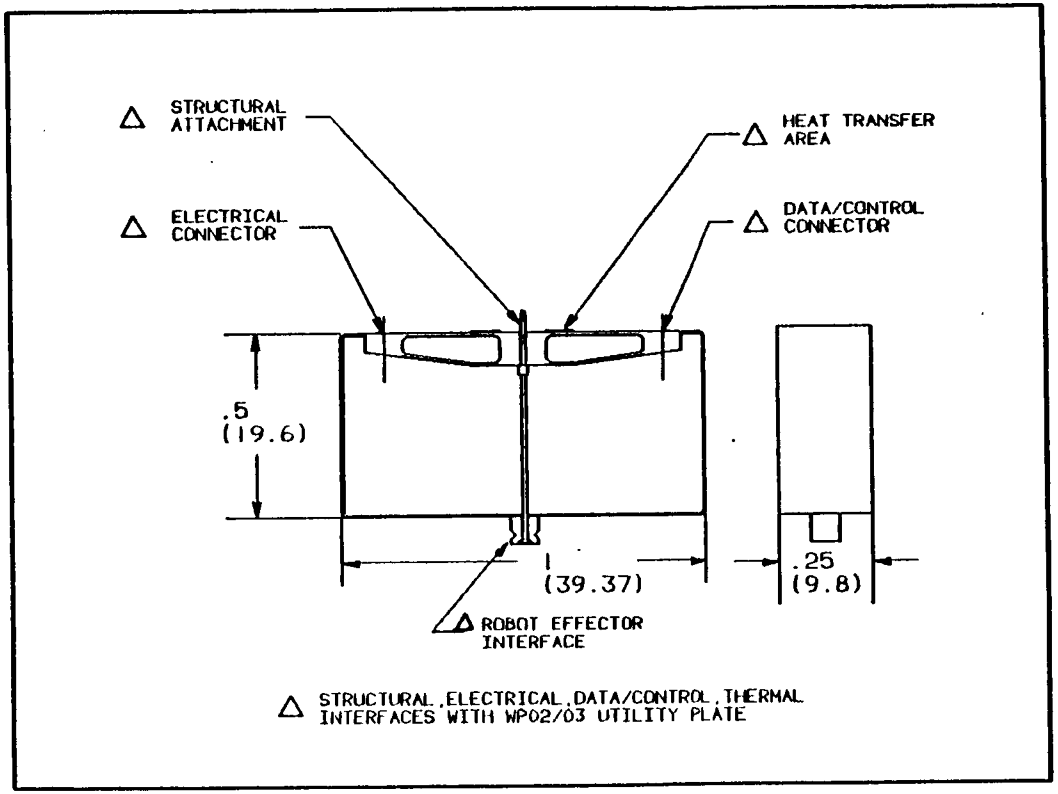

Figure 5-6. TYPICAL PMAD ORU

Operational mode information is used in conjunction with the load priority to adjust priorities as the operational mode of the station varies. For example, load priorities may change while docking operations are being conducted; or during EVA operations, IVA operations, special experiments, equipment casualties, etc.

Additional functional interfaces involve the environment at PDCA locations, as well as EMI considerations. Shadowing and operational envelope constraints of the OMV must also be considered.

Other functional interfaces with WP-02 include the following:

- Data communication with DMS
- Position information and control signals from GN&C
- Minimization of view obstructions
- Structural dynamic characteristics of power module
- Load characteristics as discussed in Section XXXX
- Possible communication and tracking data link
- Shadowing
- Contamination control

5.3.3 WP-03 INTERFACES

Major items included within WP-03 are the co-orbiting platform, polar orbiting platform, a laboratory module, attached payload accommodations, and servicing accommodations for the platforms and free-fliers.

Physical Interfaces

Physical interfaces with WP-03 involve the PDCAs applicable to this work package (see discussion in Section 5.3.2), and the equivalent list of physical interfaces with WP-02 as applied to the platforms.

Functional Interfaces

Functional interfaces with WP-03 involve the load characteristics applicable to this work package (see discussion in Section 5.3.2), and the equivalent list of functional interfaces with WP-02 as applied to the platforms.

5.3.4 Ground Support Equipment (GSE) Interfaces

The following items are typical GSE with which the EPS will interface during prelaunch processing operations. These items will require detailed verification activities similar to flight hardware.

Access GSE

Portable stands which provide access to all exterior and interior areas of the EPS hardware.

Handling GSE

Handling GSE for movement of EPS hardware and GSE. This includes air-bearing pallets with hardware support and jacking provisions, devices such as A-frame cranes and electric forklifts, overhead cranes, and slings and strongbacks for crane operations.

Protective Equipment

For all surfaces and hardware subject to scratches, tears, punctures, or impact damage.

Transportation Equipment

A transportation canister and transporter system that will provide the capability to move the elements among processing facilities while maintaining environmental controls.

Mechanical Simulators

For simulation of mechanical system interfaces not present during test, refurbishment, and maintenance verifications.

Electrical Simulators

For simulation of electrical interfaces not present during test, refurbishment, and maintenance verifications. This includes a power supply to simulate the PV and/or SD power source, a test set to simulate NSTS electrical interfaces, and equipment to simulate individual element interfaces.

Ground Data Management System (GDMS)

For ground test of the EPS element at the various support facilities. It consists of the data control, monitoring, processing, and distribution equipment and contains work stations and computer consoles with the capability to communicate with Space Station onboard systems as well as the GSE.

Loan Pool Support

For nominal and contingency operations. Includes equipment such as ohmmeters, oscilloscopes, digital voltmeters, and breakout boxes.

Voice Communications

For operations involving personnel in diverse locations.

5.3.5 NSTS Interfaces

Since the EPS will be transported into orbit by the NSTS, all elements must be totally compatible with the NSTS requirements and conform to the cargo bay interface requirements Interfaces with the NSTS include mechanical, electrical, avionics, and environmental interfaces and guidelines which are defined in ICD-2-19001, "Shuttle Orbiter/Cargo Standard Interfaces" (Reference 2).

Other interfaces with the NSTS include previsions for monitoring during flight; and deployment, heating, startup and control of the EPS until fully operational on orbit. Shadowing and contamination control must also be consistent, and power must be available during docking operations.

5.3.6 Crew Interfaces

Crew interfaces with the EPS include two main categories. One is the crew time and effort required for initial deployment and erection/construction of the EPS, repair and maintenance of EPS equipment, and replacement of ORUs. The other category includes routine operation and monitoring of the EPS. Related interfaces in this area include crew training, and operating and maintenance manuals provided for the crew's use.

5.4 INTERFACE VERIFICATION APPROACH

This section describes Rocketdyne's planned approach for verification of interfaces between Electric Power System (WP-04) and the other Space Station work packages.

5.4.1 Generic Interface Verification

Rocketdyne's work package level interface verification approach encompasses wide range of planned activities applicable to verification of physical, functional, and software interfaces. While it is important to make interface verification a consideration early in he design and development process, actual verification will, in many cases, take place at the subsystems/systems integration increment. Figures 5-7 through 5-9 illustrate the test and verification process flow for the SD, PV, and PMAD subsystems of the EPS, respectively. In these figures, interface verification is shown as an important activity during the integrated subsystem stage, and then in the form of orbiter interface verification, for the cargo launch package. Optimum location for these activities (e.g., subcontractor, Rocketdyne, LeRC, KSC) is determined on a case-by-case basis with cost and schedule as primary factors.

The interface control document drawings (ICD) depicts physical and functional work package interface engineering requirements of all items affecting the design or operation of co-functioning items. The ICD will document all necessary engineering data to:

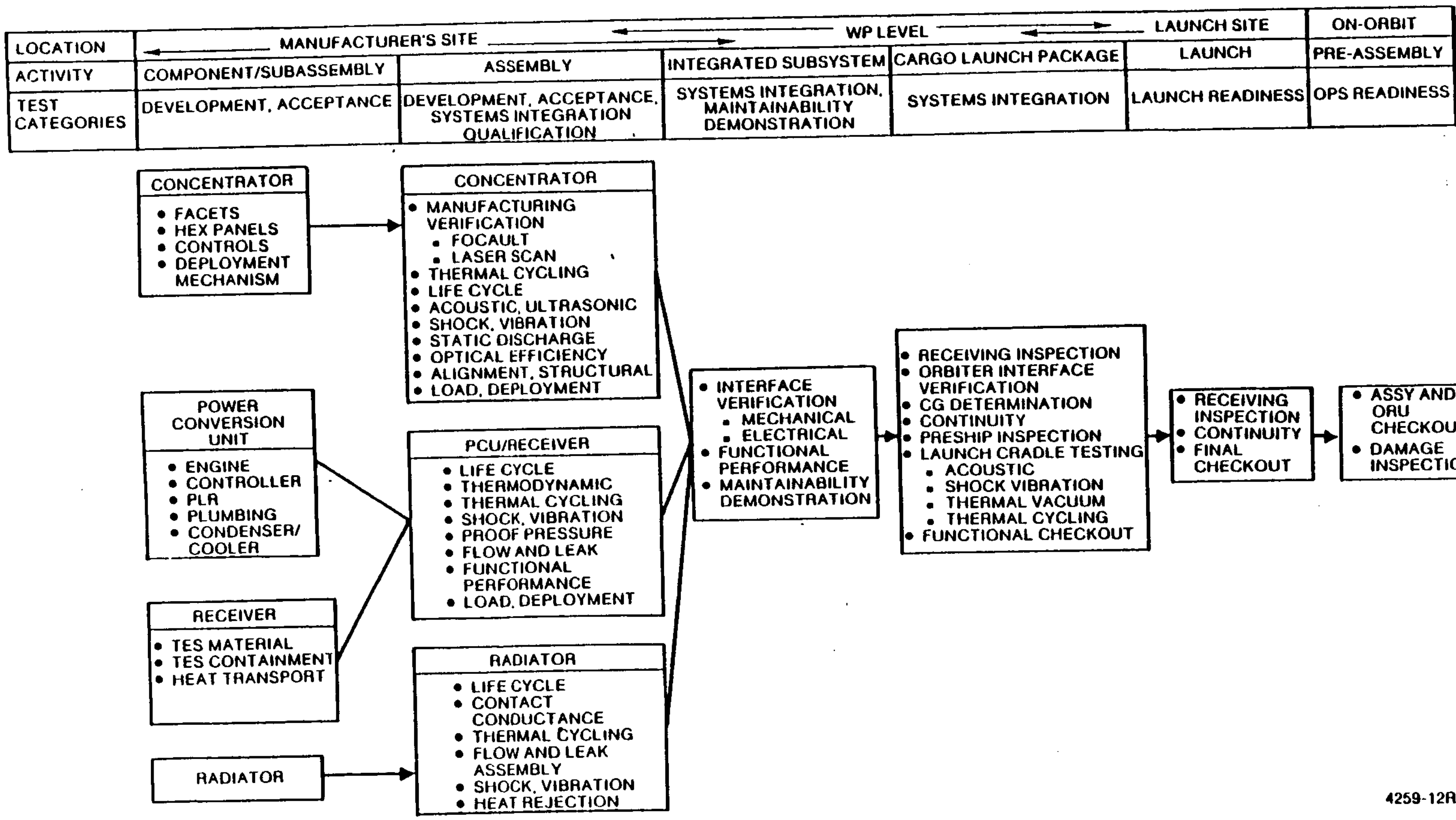

Figure 5-7. Test and Verification Flow Diagram, SD Subsystem

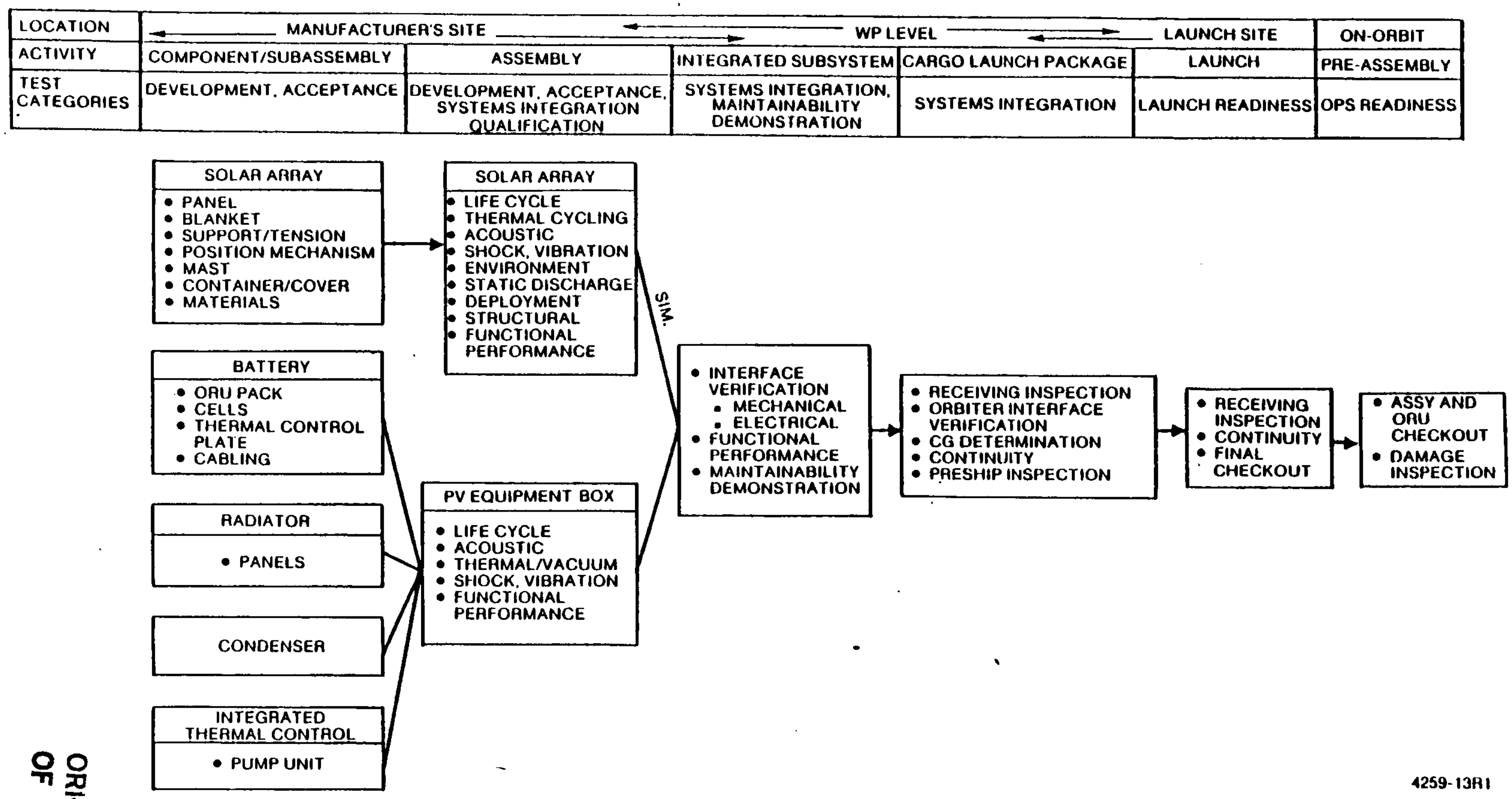

Figure 5-8. Test and Verification Flow Diagram, PV Subsystem

ORIGINAL PAGE IS OF POOR QUALITY

LOCATION	← MANUFACTURER'S SITE		← WP LEVEL →		LAUNCH SITE	ON-ORBIT
ACTIVITY	COMPONENT/SUBASSEMBLY	ASSEMBLY	INTEGRATED SUBSYSTEM	CARGO LAUNCH PACKAGE	LAUNCH	PRE-ASSEMBLY
TEST CATEGORIES	DEVELOPMENT, ACCEPTANCE	DEVELOPMENT, ACCEPTANCE, SYSTEMS INTEGRATION QUALIFICATION	SYSTEMS INTEGRATION, MAINTAINABILITY DEMONSTRATION	SYSTEMS INTEGRATION	LAUNCH READINESS	OPS READINESS

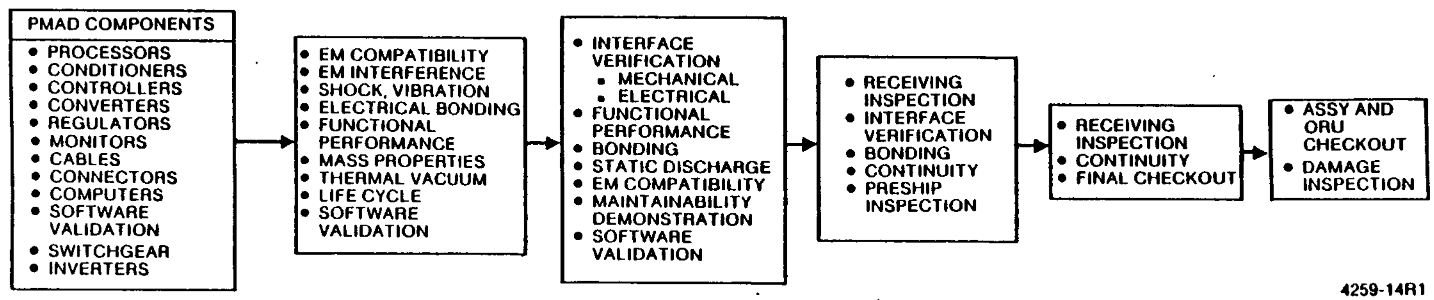

Figure 5-9. Test and Verification Flow Diagram, PMAD Subsystem

a. Establish and maintain compatibility between co-functioning items,

b. Control WP interface designs to minimize changes affecting co-functioning systems,

c. Document WP interface design baselines and changes,

d. Establish envelope and access information to ensure that co-functioning CEIs retain compatibility throughout all assembly, test, and operating conditions, and

e. Control "master gauge" dimensions and configurations where close-fit requirements exist (e.g., commonality, interchangeability, etc.).

The format and content of an ICD drawings will depend on the information required to be documented. An interface may have more than one requirement as in the case of an electrical connector where mechanical and electrical characteristics must be delineated. As the CEIs are define, WP interfaces will be grouped by type (e.g., mechanical, envelope, fluid, electrical, facilities, etc.) and once "released", and ICD will thereafter be revised only by issuance of an IFR (interface revision). This process will be controlled within Rocketdyne by an interface control project engineer, who will participate in customer/ contractor working groups, coordinate WP interface activities with associate and subcontractors, and direct internal activities throughout the development, revision, and final release of all EPS ICDs.

As a prime contractor, Rocketdyne will be responsible for the EPS ICD and upon freezing of the baseline design, will perform tolerance stackup analyses showing both sides of the interfaces we are responsible for, including those of our subcontractors. Formal verification plans will be written for subsystems/components. Prototypical hardware will be used as a design aid wherever necessary to verify development of interface designs. Formal (confirmation) fit checks will certify that hardware meets production quality and ensures that changes have no impact to the WP interfaces.

Every WP interface thus identified will be assigned expeditiously to a contractor principal manager, and all issues pertaining to that interface will be resolved prior to committing to production hardware. A control matrix will be utilized to identify the hardware and performance requirements, indicate the level and type of verification needed and status results. The verification planning process will be assessed through this methodology. Additionally, a schedule will be maintained to demonstrate how the process is supporting other dependent activities.

Maximum use of existing hardware, simulators, and common GSE will be planned. Where ground check of interfaces using flight hardware (protoflight approach) is not practical, master gauges will be used and final interface fit and operation will be verified on orbit.

Pre-test reviews will be conducted to ensure that test objectives, expected data, and safety considerations are achieved Photography will be employed as necessary to provide a permanent record for automated retrieval at any time during the DDT&E phase and operational lifetime. Post-test reviews will evaluate results versus requirements, confirm that objective have been met, and recommend any additional design/performance requirements for implementation.

5.4.2 Physical Interface Verification

Rocketdyne will define plans for conducting development, confirmation, and assembly checks. Fit checks will be utilized as necessary to develop interface designs, confirm completed designs, and verify subsequent assembly of production hardware. The plans will include identification (from ICDs) of hardware interfaces requiring fit check, and the location (e.g., subcontractor, Rocketdyne, associate contractor, LeRC, KSC) and schedule for conducting the check. Supporting requirements (hardware, facilities, manpower) will also be identified. Procedures will be generated to support the verification effort and results documented to the NASA citing action items as necessary.

Development fit checks are tools used to enhance the design and ICD effort. They normally involve only those agencies specified by the prime contractor and do not necessarily involve a review of the entire interface. Confirmation fit checks may be performed when the interfacing designs are complete, production equivalent hardware is available, and it is deemed necessary to confirm that the hardware can be satisfactorily assembled and is in conformance with applicable ICDs. Confirmation fit checks will be conducted at a site determined by hardware availability and program needs.

Early integration of verification and test goals into the total design process is essential to reduce program costs and will be a key program driver. With this in mind, the use of master gauges will be evaluated for the EPS interface verification task. Controlled by tool drawings, a master gauge will be manufactured, where deemed necessary, by the principal contractor for the interface and copies made for other contractors and NASA for locating common equipment (e.g., PMAD load centers) on other work packages and co-designing close-fit hardware (e.g., beta joint, roll rings, etc.). The gauges will be calibrated and temperature controlled or compensated for critical tolerance control. Every common WP interface attach point can thus be controlled and fabricated with maximum utilization of existing designs and a concurrent reduction of risk that part-to-part deviations normally imply.

Physical interfaces of PMAD components include connections to the support systems furnished by other work packages and the mounting of components to the station structure. The use of standard electrical connectors at sites will enable a one-time verification of this interface.

After verification for launch, interfaces will be controlled by ICD change process. If changed or disconnected, reverification will be performed.

5.4.3 Functional Interface Verification

A test and verification flow control diagram will be generated to ensure tht all major components/subassemblies are schedule for test in a logical, meaningful, and reasonable manner. Functional interfaces will be verified through ground testing and will provide progressive verification from piece part through subsystem and integrated system testing. System test requirements which are not intended to verify WP interfaces will also be reviewed to ensure what the integrity of previously verified hardware/software is not compromised. Only on completion of scheduled tasks will a subsystem be verified as ready for integration into its next highest program element or module.

Functional interface verification is critical to assure that the overall EPS will operate in a predictable and reliable manner. Load characteristics verification is required for all electrical loads. The load characteristics required will be those found at the point of PMAD control which is the remote power controller (RPC) output. The RPC is the functional interface point between each load and the EPS. The central power management process functional interface with the WP-02 DMS is discussion in Section 5.4.4.

The use of simulators and test beds will be an important part of functional interface verification in order to compensate for differences in delivery schedules and manufacturing/test sites. These will enable work package level interfaces to be verified before actual function can be demonstrated at the system integration site.

Test beds will be used to simulate the EPS. These test beds will provide a hardware test environment where interfaces will be verified. The components of the test bed will include a representative sample of all PMAD components.

WP-04 will provide a simulator that will generate station type electrical power. The simulator will provide the user with power at the selected distribution frequency and voltage. User connection to this simulator will be made through a remote power controller (RPC) and electrical connectors. The interfaces will be generic to those used on the station. This simulator will allow different users and work packages to verify that their WP-04 interfaces are correct and will provide the means for powering Space Station elements for ground test. The design will include provisions for ground support equipment connections at critical locations so that the system can be easily verified and worked on if necessary.

WP-04 will require load characteristic simulation from all other work packages and specifically from WP-02 a simulator that simulates the station DMS (including the bus). The DMS simulator will be used to verify the power management processor/DMS interface along with the crew interface. A test set will also be used to support verification of NSTS electrical interfaces.

5.4.4 Software Interface Verification

In the EPS reference configuration, all EPS software is internal to the various EPS processors. The only external software interface the EPS has with the Space Station is through the power management controller (PMC) and the station DMS/OMS communication bus. The EPS will be controlled internally by a hierarchal set of software processor, the top level of which is the PMC.

The PMC will respond to commands and overrides from the station OMS,and will accept selected data inputs including priority lists, and pointing and tracking information.

The PMC interface will not carry any commands from the EPS to station DMS/OMS. However, upon request from OMS, EPS will provide data on status of EPS, available power, maintenance recommendations, and power system topology. The functional interface between the station data management system (DMS) and the EPS central processor is so software intensive that its verification should be approached from a software standpoint. Each type of message between the DMS and EPS must be examined as a software interface, on an end-to-end basis. If the DMS simulator includes all these messages, the verification will be by test. The principal tool for verification of the EPS external interface is the DMS simulator which will be required by start of PDR (8/88).

Test and verification of software will occur throughout the software development process. Specific plans and procedures will be defined in general at the beginning of the development effort, and details for each phase will be generated as a part of the previous phase. Plans for this effort, based on experience in other projects, and our present understanding of requirements and likely software development environment, are discussed below.

The first phase of the effort will consist of generating detailed requirements documents for each of the software "programs." (The term "program", as used herein, is that set of software which runs on one processor.) This actually a systems synthesis/analysis task, not a programming task. Verification that the set of processor programs, as defined by the requirements document, will work together to accomplish the overall job will be done by system simulation. The depth to which this simulation/verification can or will be done, will be determined by the simulation tools and other resources available at the time. It is never possible or cost-effective to completely simulate a complex system, so systems engineering judgment will be used to define which parts of the system will be simulated, and to what degree.

After the requirements for each software program are defined, the design phase for that program can proceed. This phase consists of determining how each program is to be structured and subdivided into individual, small software components. (Variously called modules, subroutines, functions, flow segments, etc.) The output of this phase will consist of the structure charts or lists which show how the program fits together and a preliminary design for each component in a program design language (PDL). If the software development environment includes an Ada based PDL, it will be used in the design. Otherwise, a natural language PDL will be used. Normally, the verification of the software design against the requirements consists of reviews by programming personnel not involved in the design. This will be the procedure followed for the PMAD software development, unless the SDE includes an executable type of PDL. Prototype code may be developed and tested to support parts of the design process and used in the review process.

Following the completion and review (or test) of the design of each component, coding can begin for that component. (Normally, the design of an entire program will be completed before coding starts on any part, but sometimes parts which have clear interfaces to other parts can start early.) As each component is coded in an HOL (probably Ada), it will be compiled and tested in a test environment. This environment will consist of test software which includes stimuli for the component under test, and analysis of the component's performance. These stimuli will consist of both normal and abnormal inputs, to measure the component's exception handling capabilities.

As individual components are completed, and assembled into more complex functions, these collections of components will be tested in the same manner Tests to this point are conducted entirely in the software development simulation environment. When collections develop to the point where they can be installed in target hardware, the test program will expand to that type of environment. This is when errors in the software requirements are first likely to be discovered and changes made thereto.

During the coding and testing process described above, reviews of the code itself, and the test results, will be conducted with personnel not personally involved in the coding. By the time the coding of each of the programs is completed,its components will have been tested and reviewed many times, at many levels.

As programs are completed, they will be integrated into an overall test bed. This test bed will include hardware which is representative of that which the software will interface with and control. It will include a hardware and software test environment which will permit setting up test conditions for the software and monitor software performance. At this point, both the software requirements and the software itself will be subjected to thorough testing to a formal plan.

5.4.5 Ground vs On-orbit Verification

Interface verification is an important activity which must be performed all though the test and verification process. It represents an integral part of development testing, acceptance and certification testing, systems integration testing, launch readiness testing, and finally on-orbit verification.

Maximum interface verification will be performed on the ground. This will enable problems to be detected at the earliest possible time and corrections to be made for the lowest possible cost and risk. Physical and functional hardware and software simulators or physical mock-ups will be used to test interfaces before actual fit and function can be verified at the system integration site. These simulators will be necessary to compensate for differences in delivery schedules, manufacturing/test sites, and assembly procedures and are particularly critical for interface verification at the work package level. All work package level physical and functional interfaces will be demonstrated as compatible and functional before launch to as great a degree as practical.

On-orbit verification is verification on the Space Station itself. Interface verification on-orbit is expensive in terms of crew time and training requirements and may also have crew safety implications. However, final interface verification of the end-to-end power system will be done on-orbit since it does not appear cost effective or practical for the entire Space Station to be assembled and tested on the ground.

The choice of site for interface verification activities which can be performed either on-orbit or on the ground (e.g., LC and payload) depends on a variety of considerations. These include cost, crew time, crew training requirements, safety of crew and equipment, installed BITE capability, GSE requirements, and schedule.

5.4.6 Prototype vs Protoflight

The protoflight concept involves the use of flight hardware for qualification testing in lieu of a dedicated test article (prototype). This approach is applicable to few WP-04 elements at this time, however its use will be maximized wherever cost effective and practical.

Utilization of flight hardware for some ground testing will enable physical and functional interfaces of WP-04 with other work packages to be verified with high reliability and accuracy. This will be accomplished best when both actual Space Station mating flight elements are available at the system integration site, and alternatively, when one is available and a simulator or physical mock-up is employed for the other.

5.4.7 Growth Interface Verification

Interface verification of growth hardware after the IOC Space Station is on orbit adds a new set of considerations to the interface verification approach. With the Space Station on orbit before some growth elements and payloads are fabricated, interface verification becomes more of an operational challenge. Dependence upon simulators and physical mock-ups will increase since growth hardware may not have access to the Space Station elements with which they will interface on orbit.

The complexity of growth interface verification depends to a large extent on whether growth is being accomplished by replication of existing IOC elements, or if new and different hardware is being used. Commonality of growth items with IOC items will ease this task by allowing the use of existing facilities, drawings, procedures, etc. Strict configuration control and management will ensure that IOC components are not modified in a way that could cause interface problems during the growth phase. All modifications will be verified against the same simulators, mockups, or master tools as the initial hardware which is already on orbit to assure proper interface; and, when changes are made, the resulting hardware must again be verified by these means to assure configuration control.

Hardware to be delivered to an on-orbit Space Station will undergo physical and functional interface verification with the same ground based simulators, mock-ups, or master tools that were used to ensure proper fit and function of the initial station equipment.

5.4.8 Use of Built-in Test Equipment (BITE)

The use of BITE as an EPS verification tools is an important consideration for on-orbit verification, and in providing a measure of the component's health during its operating life. The degree of BITE utilized on the Space Station will be determined on a case-by-case basis with one important factor being the adoption of a man-tended Space Station approach.

The PMAD subsystem will include on-orbit self-testing that will provide an operator with the ability to verify the operational readiness of any component. When a malfunction occurs, this feature will alert the operator of the need for ORU maintenance or replacement.

5.5 REFERENCES

Information from the following documents has been used in preparation of this plan.

1. NASA-LeRC RUR 2-4 EMS Data, Verification and Checkout, 20 September 1985.

2. Shuttle Orbiter/Cargo Standard Interfaces, J8400005 JSC 07700, Volume XIV, Attachment I (ICD 2-19001).

3. Software Development, Test and Verification Plan, DR-16, Rocketdyne, 19 December 1985.

4. Space Station Prelaunch Operations Plan (Preliminary), JSC 30202, 19 August 1985.

5. Systems Test and Verification Plan Content Guide, J8400082, 15 June 1984.

6. Time-Phased SE&I Study Products DR-19, DP 4.3, Rocketdyne RI/RD 85-194-2, 3 October 1985.

7. Time-Phased SE&I Study Products DR-19, DP 4.4, Rocketdyne RI/RD 85-194-3, 19 November 1985.

8. Test Requirements for Space Vehicles MIL-STD-1540B.

9. Space Station Operations Process Requirement Document, JSC 30286

10. Space Station Operations Plan, JSC 30201

11. Master Verification Process Requirements (TBD)

12. Verification Integration Processes Requirements (TBD)

6.0 CUSTOMER ACCOMMODATIONS

6.1 DESIGN APPROACH

Work Package 04 has the responsibility of providing utility power to all customers (housekeeping loads and payloads). Details of Work package 04 Electrical power system design is presented in section 2.3.

All electrical loads are served from the Power Distribution and Control Assemblies (PDCA) which are located throughout the station. Each PDCA contains Remote Power Controllers (RPC) that function as the electrical interface with each load. Three sized (75 amp, 25 amp and 5 amp) of RPCs are provided to the user. Connection to more than one RPC is required for fault tolerant operation. The user can chose to connect the load as a critical load which requires three RPCs, essential load which requires two RPCs or as a non-essential load requiring only one RPC.

Work Package 04 will supply power to Work Package 02 utility ports and Work Package 01 equipment racks as well as Work Package 03 utility ports. Utility ports and rack locations will be determined by other work packages.

6.2 RESOURCES

Work Package 04 generates and distributes power resources for the Space Station and Platform. Table 6.2-1 list EPS design considerations and design approaches used to accommodate the customer.

EPS/Customer interface is at the PDCA. All PDCAs on the Space Station and Platform delivers utility power of the same voltage, frequency and other characteristics shown in Table 6.2-2. This allows payloads to be moved from one station or platform location to another without modification. There are 22 PDCAs located on the Space Station as shown in Figure 6.2-1. A total of ten PDCAs are located throughout the truss structure at regular intervals to support truss mounted loads. Loads within manned modules are serviced by 12

CONSIDERATION	APPROACH
CUSTOMER SECURITY	EPS DESIGN DOES NOT POSE A CUSTOMER SECURITY PROBLEM
EASE OF PAYLOAD INTEGRATION	WP-04 PROVIDES USER LOAD CONVERTERS AND A 20 kHz 208 VAC GSE POWER SOURCE FOR PRE-ORBIT PAYLOAD CHECKOUT
EASE OF PAYLOAD CONFIGURATION	EPS PAYLOAD DATA BASE CAN BE UPDATED AT WILL VIA DMS COMMANDS. 802 CUSTOMER CONNECTION POINTS LOCATED THROUGHOUT THE STATION
EASE OF PAYLOAD SERVICING	POWER CAN BE DE-ENERGIZED TO PAYLOAD AT WILL BY THE EPS VIA DMS COMMANDS
EASE OF PAYLOAD PACKAGING	WP-04 NOT INVOLVED IN PAYLOAD PACKAGING
DEGREE OF TRANSPARENCE OF PAYLOAD OPERATIONS	CUSTOMER OPERATION IS COMPLETELY INDEPENDENT OF PAYLOAD OPERATION
INDEPENDENCE OF PAYLOAD OPERATION	AS LONG AS LOAD IS ALLOCATED POWER BY THE DMS (NOT A PART OF WP-04), CUSTOMER PAYLOADS OPERATE IN COMPLETE INDEPENDENCE OF THE EPS.
RESOURCES PROVIDED	SEE TABLE 6.2-2
PAYLOAD ENVIRONMENT	WP-04 DOES NOT EFFECT PAYLOAD ENVIRONMENTS

DESIGN CONSIDERATIONS
TABLE 6.2-1

- POWER SOURCES (STATION)
  - 75 kW AVERAGE POWER AT IOC
  - 100 kW PEAK POWER AT IOC
  - 300 kW AVERAGE POWER AT GROWTH
  - 350 kW PEAK POWER AT GROWTH
- POWER SOURCES (PLATFORM)
  - 8 kW AVERAGE POWER AT IOC
  - 18 kW PEAK POWER AT IOC
  - 24 kW AVERAGE POWER AT GROWTH
  - 34 kW PEAK POWER AT GROWTH
- UTILITY POWER CHARACTERISTICS
  - PDCA MAXIMUM POWER (25) kWe
  - FREQUENCY 20 kHz ± 2%
  - VOLTAGE 208 VRMS, SINGLE PHASE ± 2.5%
  - MINIMUM POWER FACTOR .9
  - HARMONIC DISTORTION < 3% TOTAL
  - VOLTAGE DROPOUT DURATION, 50 MSEC. MAXIMUM
  - TRANSIENT VOLTAGE, ± 10% MAXIMUM FOR 250 msec.
  - GROUND LINE CURRENT, < 15 MA NOMINAL (FULL LOAD)

POWER RESOURCES TO CUSTOMERS

TABLE 6.2-2

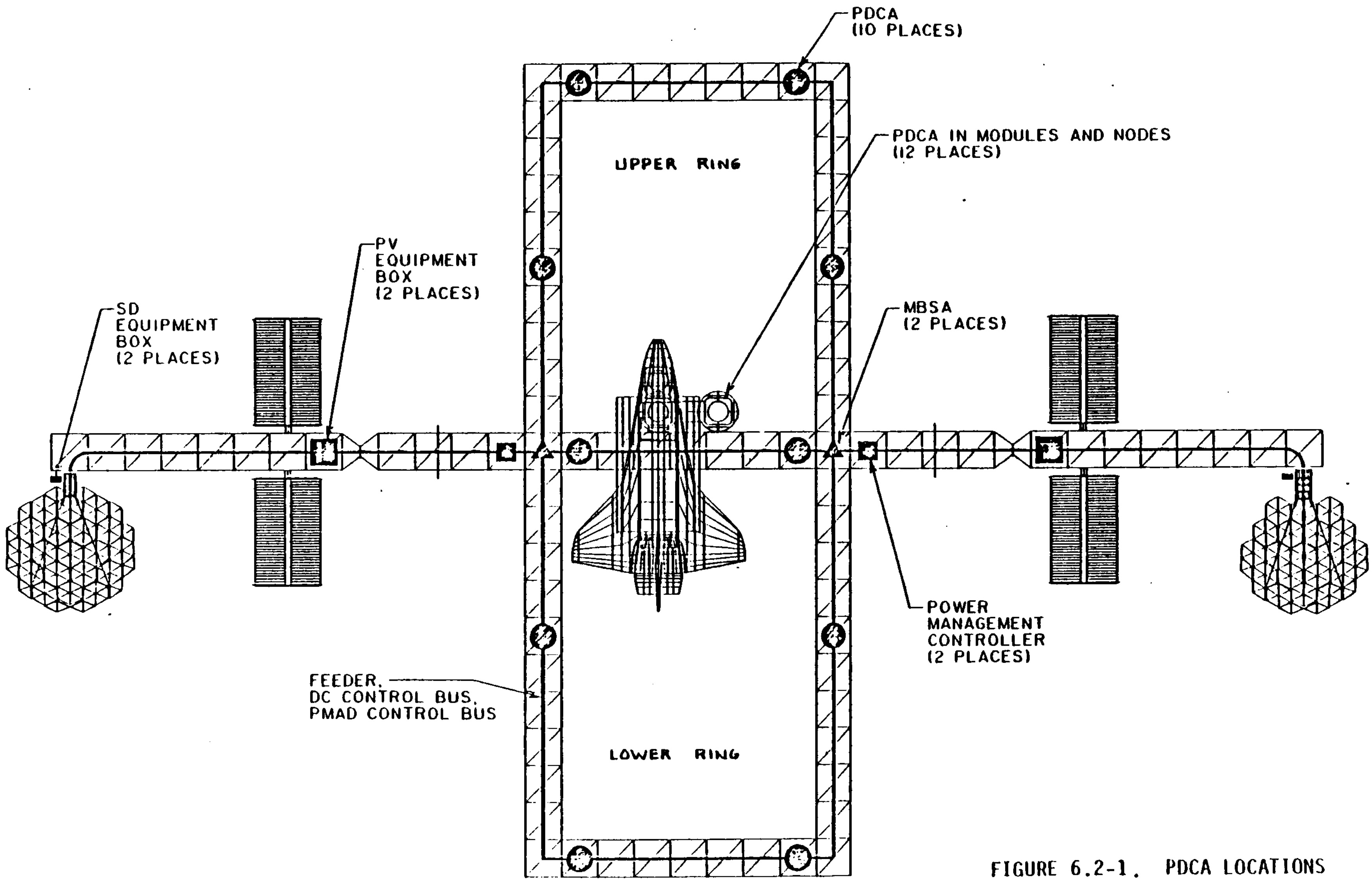

FIGURE 6.2-1. PDCA LOCATIONS

Rockwell International Corporation
Rocketdyne Division
SPACE STATION EPS COMPONENT LOCATIONS
C 02602 7R070021

PDCAs. Table 6.2-3 lists the number of PDCAs serving a feeder segment or pressurized module. This table also includes the number of customer connections (RPC interfaces) and maximum power capacity.

The power management and distribution system of the platform is nearly identical to that of the station. Because of the platforms' smaller size, only two PDCAs (one housekeeping and one payload) are used. Electrically and mechanically, the platform's user interfaces maintain a high degree of commonality with that of the station. Payloads are attached to PMAD system the same way as on the station.

Total power for both co-orbit and polar orbit platforms are shown in Table 6.2-2. Of the total power 3 kW is reserved for housekeeping and not available to the customer.

FEEDER/MODULE	NUMBER OF PDCA	NUMBER OF CUSTOMERS CONNECTIONS	MAXIMUM CAPACITY (kW)
UPPER RING	5	180	25
LOWER RING	5	180	25
HAB MODULE	5	180	50
LAB MODULE	5	180	50
JEM MODULE	0	4	50
ESA MODULE	0	4	50
PRESSURIZED PAYLOAD	0	2	50
NODES	2	72	50
TOTALS	22 (3)	802 (2)	(2)

NOTE 1 Each PDCA contains: (10) 5 AMP RPC, (8) 25 AMP RPC and (18) 75 AMP RPC.

NOTE 2 Total station power capacity is limited to 75 kW at any one time.

NOTE 3 See Figures 6.1-1 for PDCA locations.

UTILITY POWER CONNECTIONS

TABLE 6.2-3

6.3 LOAD CONVERTERS

A review of the Space Station Mission Data Base and NASA Data Book indicates that customer loads will fall into ten general categories of voltage and power levels. Work Package 04 will design, qualify, and produce a family of ten load converters which will satisfy most customer needs. For commonality and ease of integration, all space station customers can use this family of load converters thus lowering payload development costs. Table 6.3-1 lists the ten load converts along with some design data.

LOAD CONVERTER	VOLTAGE	FREQ (Hz)	PHASE	POWER (watts)	REG (%)	MASS (lbs)	LENGTH (in)	WIDTH (in)	HEIGTH (in)	THERMAL (watts)	EFF (%)	LOAD DESCRIPTION
LOAD CONVERTER #1	120	400	3	200	5	12	10	4	4	20	90	LIGHTS, SMALL MOTORS
LOAD CONVERTER #2	208	400	3	500	5	25	10	5	5	45	91	PUMPS, MOTORS
LOAD CONVERTER #3	TBD	VAR	1	1000	10	40	15	6	6	80	92	INDUCTION HEATING DEVICES
LOAD CONVERTER #4	TBD	VAR	1	500	10	25	10	6	6	45	91	HEATING DEVICES
LOAD CONVERTER #5	5	DC	-	200	2	5	5	3	3	40	80	ELECTRICAL PROCESSORS AND CONTROLS
LOAD CONVERTER #6	+/-15	DC	-	1000	2	40	15	5	5	150	85	ELECTRICAL/INSTRUMENTATION DEVICES
LOAD CONVERTER #7	50	DC	-	500	5	20	10	5	5	85	80	CONTROLS, DEVICES
LOAD CONVERTER #8	28	DC	-	1000	10	40	15	5	5	150	85	CRITICAL DEVICES
LOAD CONVERTER #9	150	DC	-	200	2	10	6	3	3	30	85	BATTERY PROCESSES
LOAD CONVERTER #10	400	DC	-	500	5	5	10	5	5	70	86	TRANSMITTERS

LOAD CONVERTERS

TABLE 6.3-1

6.4 INTERFACE REQUIREMENTS

Some Work Package 04 customer interfaces have been selected for discussion in this section. These interfaces are EPS power characteristics and customer load characteristics. Exact details of the EPS/customer interface are covered in an Interface Control Document (ICD). A preliminary list of parameters covered in this ICD are shown in Table 6.4.1. As the design matures, values, specifications and part numbers will be added to the control document.

TABLE 6.4-1

PRELIMINARY WP-04/CUSTOMER INTERFACE REQUIREMENTS

EPS POWER CHARACTERISTICS:	Voltage Frequency Phase Polarity Power quality EMI/EMC
CUSTOMER LOAD CHARACTERISTICS:	Power Power Factor Impedance EMI/EMC Grounding Priority Classification (crew critical, station critical, payload critical, deferrable and non-essential) Load Location
MECHANICAL	Power Connector Pin Functions Wire Gauge Size

7.0 DESIGN TRADE OFF STUDIES

7.1 PEAKING SPLIT

A trade study was performed to determine the peaking split between PV and SD, and specifically to compare the inherent total peaking capability with proportionally shared peaking (i.e., 33% each).

As a basis for this study the current IOC module sizes were used, which result in an IOC station with 23.5 kw PV and 51.5 kw SD. Growth would be accomplished by the addition of replicated SD modules, adding 51.5 kw of SD per growth step. The peaking requirements are 100 kw during IOC and 350 kw for the growth station (300 kw nominal).

As currently designed, both the PV and the SD modules have certain inherent peaking capabilities, that is, they can provide some peaking with little or no additional substantive cost or mass penalty. These inherent peaking capabilities have been estimated and are tabulated in Table 7.1-1.

TABLE 7.1-1
INHERENT PEAKING CAPABILITY

	Nominal IOC Power (kw)	Inherent Peaking Capability (%)	IOC Peaking Capability (kw)
PV Module	11.75	81	21.25
SD Module (ORC)	25.75	15	29.5
SD Module (CBC)	25.75	15	29.5

Table 7.1-2 illustrates the peaking capability of the IOC and growth stations for both the inherent peaking concept and the proportional peaking concept. Two PV modules and two SD modules are used on the IOC station to provide the nominal 75 kw power requirement. The growth station would contain 12 SD modules (332 kw total net power) in order to exceed the 300 kw nominal growth requirement.

TABLE 7.1-2
INHERENT AND PROPORTIONAL PEAKING COMPARISON

	Inherent Peaking (kw)		Proportional Peaking (kw)	
	IOC	Growth	IOC	Growth
PV Power	42.5	42.5	31.3	31.3
SD Power (ORC)	59.0	354.0	68.7	412.0
SD Power (CBC)	59.0	354.0	68.7	412.0
TOTAL EPS	101.5	396.5	100.0	443.3

Note from Table 7.1-2 that utilization of the inherent peak power capability of the PV and SD modules exceeds the peaking requirements of 100 kW at IOC and 350 kW at growth.

Table 7.1-3 provides the estimated cost and mass differential for proportional peaking. Both the CBC and ORC concepts result in cost and mass penalties relative to the inherent peaking base.

Based on these results it is recommended that the inherent peaking capability of the PV and SD modules be utilized to meet the station IOC and growth peaking requirements.

TABLE 7.1-3
INHERENT AND PROPORTIONAL PEAKING COST AND MASS COMPARISON

OPTION	INHERENT SD AND PV PEAKING	PROPORTIONAL SD PEAKING	
		CBC	ORC
IOC COST (1987 $M)			
SD	BASE	+18.6	+ 5.4
PV	BASE	BASE	BASE
TOTAL	BASE	+18.6	+ 5.4
GROWTH COST (1987 $M)			
SD	BASE	+45.6	+14.4
PV	BASE	BASE	BASE
TOTAL	BASE	+45.6	+14.4
IOC MASS (LBS)	BASE	+2900	+1000

7.2 GIMBAL JOINTS

7.2.1 Introduction

The beta gimbal joints, used on the Space Station, and the alpha gimbal joint on the platforms, all perform the same function; positioning of the PV solar arrays , and the SD solar concentrator. This analysis supports and documents the selection of the gimbal joint design. A description of our gimbal joint design concept is contained in Section 2.2.6.

7.2.2 Design Aspect Requiring Trade-Off Study

The design aspect requiring a trade-off study was the degree of commonality among the station PV joints, SD beta joints, and the platform alpha joint.

Elements of design such as the number of main bearings in the beta joints, and the type of joint drive motors, were resolved as part of the design process.

7.2.3 Degree of Commonality Among the Station Beta Joints (PV & SD) and the Platform Alpha Joint

The following approaches were considered as to the possible degree of commonality.

A) Individual tailored design for the station SD, the station PV and the platform.

B) Commonality of joints for the station SD & PV and a special one for the platform.

C) Commonality among the station PV, SD and the platform joints.

Evaluation of the alternatives or options with respect to the various relevant criteria is summarized in Tables 7.2-1 through 7.2-3.

Each element was given equal weight with a 10 rating as maximum. The rating is relative, e.g., the combination with maximum engineering effort received zero, and that with minimum engineering effort received 10. Engineering judgement was utilized in considering the elements which are affected by the quantities of a given type of joint, e.g., procurement efforts.

The element of cost is implicit in all the elements listed. Explicit cost data was not available and is not be expected to change the order of the recommended alternatives.

Table 7.2-4 summarizes the comparison of the approaches evaluated.

TABLE 7.2-1

INDIVIDUALLY TAILORED DESIGN FOR THE STATION SD, PV AND THE PLATFORM (OPTION A)

ELEMENT	COMMENT	RATING
Design Effort	This will require the maximum design effort.	0
Procurement	While components for these different joints are required, still there are 4 each for PV, and 12 each for SD on the station.	7
Manufacturing	Same as for Procurement.	7
Assembly and Testing	Different tools and fixtures are required, although some commonality is expected.	3
Packaging for the NSTS	Three different types are required.	0
Weight	With individual design the weight is kept to a minimum.	10
Reliability	With oversizing and weight kept to a minimum some relative loss of reliability is expected.	5
Maintainability (EVA)	A simple design is expected.	10
Spares (ORU)	Requires the maximum variety and quantity of spares. However, consideration should be given to the actual number of each joint.	3
Interfacing	Requires the maximum variety. Still considerations are given to the number of each joint.	7
	TOTAL	52/100

TABLE 7.2-2

COMMONALITY OF JOINTS FOR THE STATION SD & PV AND A SPECIAL ONE FOR THE PLATFORM - (OPTION B)

ELEMENT	COMMENT	RATING
Design Effort	Two different designs are required.	5
Procurement	16 out of 18 joints have the same elements.	9
Manufacturing	Some slight difference between the Station PV & SD.	8
Assembly and	Number of different tools and fixtures is low.	6
Packaging for the NSTS	Two different package types are required.	5
Weight	Some weight growth is expected on the station and the platform joint.	6
Reliability	Reliability should increase with the slight overdesign of the station PV joint.	7
Maintainability (EVA)	Same as Option A.	10
Spares (ORU)	The number of spares are low.	8
Interfacing Hardware	There is a high interface uniformity	9
	TOTAL	73/100

TABLE 7.2-3

COMMONALITY AMONG THE STATION SD & PV AND THE PLATFORM JOINTS
(OPTION C)

ELEMENT	COMMENT	RATING
Design Effort	Single design is required with some minor differences.	9
Procurement	Single - uniform effort.	10
Manufacturing	Only slight differences.	9
Assembly and Testing	Uniform tools and fixtures with slight differences.	9
Packaging for the NSTS	Uniformity maximized - some differences still exist	8
Weight	The overall weight is highest.	2
Reliability	With overdesign for the Station PV and the platform, the reliability improves.	8
Maintainability (EVA)	Same as Option A.	10
Spares (ORU)	The number of spares are the minimum possible.	10
Interfacing Hardware	Maximum uniformity, with slight differences.	9
	TOTAL	85/100

TABLE 7.2-4

COMPARISON MATRIX

APPROACH	PRO	CON	RATING
A. Individually designed joints	Minimum weight.	Everything else particularly ORU spares are the maximum.	52
B. Station beta joint same, platform joint unique	All elements are above average.	No one particular elements.	73
C. All joints are practically the same	Spares minimized reliability maximized. All advantages of commonality are maximized.	Weight will be the highest.	85

The advantage of approach B over approach A is pronounced. The case for commonality is not as strong when comparing approaches C and B, for the following reasons:

1) There are six beta joints on the IOC station, which after growth expand to sixteen, of which twelve are for the SD and four are for the PV. There are only two joints on a platform. Hence, commonality of the beta joints on the station affects 22% of the joints, while the alpha joint on a platform affects only 11% of the joints.

2) The contribution of platform joints to the overall commonality is balanced by the expected increase in weight of the platform joint. This is not the case on the station, i.e., no significant weight penalty to the station PV beta joints, due to its commonality with the SD joint is expected.

Roll rings were included in all joints. A case can be made for flexible cables in the beta joints for the station. However, it was assumed that roll ring reliability will be equal to or better than flexible cable reliability. Furthermore, it was assumed that less spares are required, and that the ORU maintenance for the roll rings is no more difficult than that for flexible cables.

It is recommended that: 1) full commonality for the station beta joints and the platform alpha joint be employed; 2) the platform alpha joint design and commonality be reevaluated when the exact alpha joint to platform interface is known; and, 3) roll rings be utilized throughout the station and platform joints.

7.3 CONCENTRATOR STRUCTURE TRADE STUDY

7.3.1 Summary

The preliminary structure dynamics design analysis, reported in the June Issue of DR02, in support of the Solar Dynamic (SD) concentrator, interface structure and fine pointing controls was updated. The objective of the updated analysis was to obtain greater stiffness and better optical qualities for a novel concentrator fine pointing mechanism and interface structure concept. The specific goal was to evaluate the new configuration with respect to mass and coupled structural vibration modes. Modal frequency constraints ($\geq$ 1 Hz) were derived from the desire to separate structural modes from the fine pointing control loop bandwidth (0.5Hz) by a factor of two, to preclude control/structure interaction. It was concluded from the results of this study that the new fine pointing mechanism/interface structure configuration is both low in mass and sufficiently rigid to effectively avoid modal frequencies below one Hertz.

7.3.2 Reflective Surface Configuration

The reflective surface configuration reported in the June issue of DR02 consisted of a hex-truss modular construction using graphite-epoxy support beams for mirrored facets. For convenience, each hex-truss was modeled as 12 major beam elements, with the weight of hinges, latches, beam elements and facets lumped at the various circumferential and interior nodes. The actual reflective surface configuration previously employed 42 graphite-epoxy beam elements per hex-truss, with each major beam element represented by two beams, and 18 additional interior support beams. The additional interior beam elements were modeled as mass at the nodes, and a 50% increase in element bending stiffness was assumed. Adjacent hex truss modules were represented as double beam elements joined at the vertices.

No structural consideration was given to latch and hinge installations. Final mass for the previous isolated structure model of the reflective surface subassembly was 532 kg (1174 lbs).

Since the June submittal of DR02, the design of the reflective surface has been simplified and the mass previously analyzed significantly increased. From the point of view of effectively modeling the reflective surface, independent models of the isolated reflective surface with the current configuration and mass, showed a significant increase in overall stiffness, in spite of the increase in mass. The increased stiffness was attributed to an improvement in the reflective surface support structure*. The current reflective surface configuration, described in section 2.2.3, now closely matches the previous model. Since the isolated reflector model of the current configuration, including the current mass, showed no degradation in stiffness, the reflective surface model was not changed.

7.3.3 Interface Structure And Strut Configuration

Figure 7.3-1 depicts the previous and current integrated interface structure, strut and reflective surface assembly configuration. The previous interface and strut structure is represented by a three-point space frame network, made up of a back-up truss, main mast, supporting struts and "T-brace" interface structure assembly for attachment to the beta joint and mounting of the PCU equipment. The current configuration of these items is represented by a prismatic truss space frame, reinforced by a triangular frame, made up of equal diameter struts; and a double ring-gimbal fine pointing mechanism, attached to a space frame interface structure superstructure mounted to a base plate. In both the previous and current configurations, all frame elements are of filament-wound graphite-epoxy construction. Frame element sizing is summarized in Table 7.3-1

In the previous configuration, the reflector-end of each strut and the main mast was allowed to rotate in two directions but prevented from free-torsional motion, so as to simulate the existence of actuators and universal joints to

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* JA6763 "Solar Dynamic Reflector Final Report for Space Station Work Package 4 Phase B Program", 30 September, 1986, Harris Corporation, Melbourne, FLA.

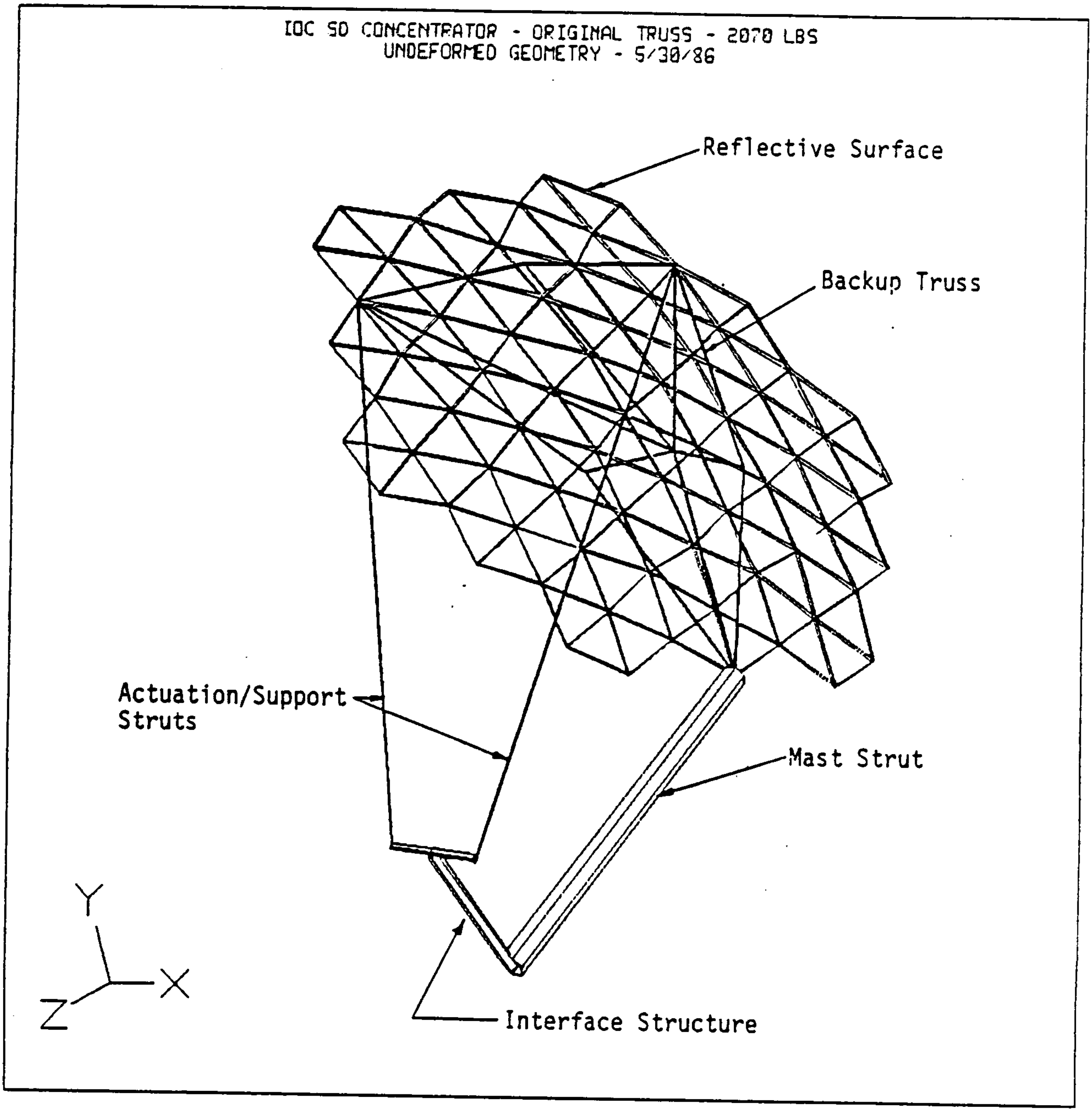


Figure 7.3-1a - Previous Concentrator Model Configuration

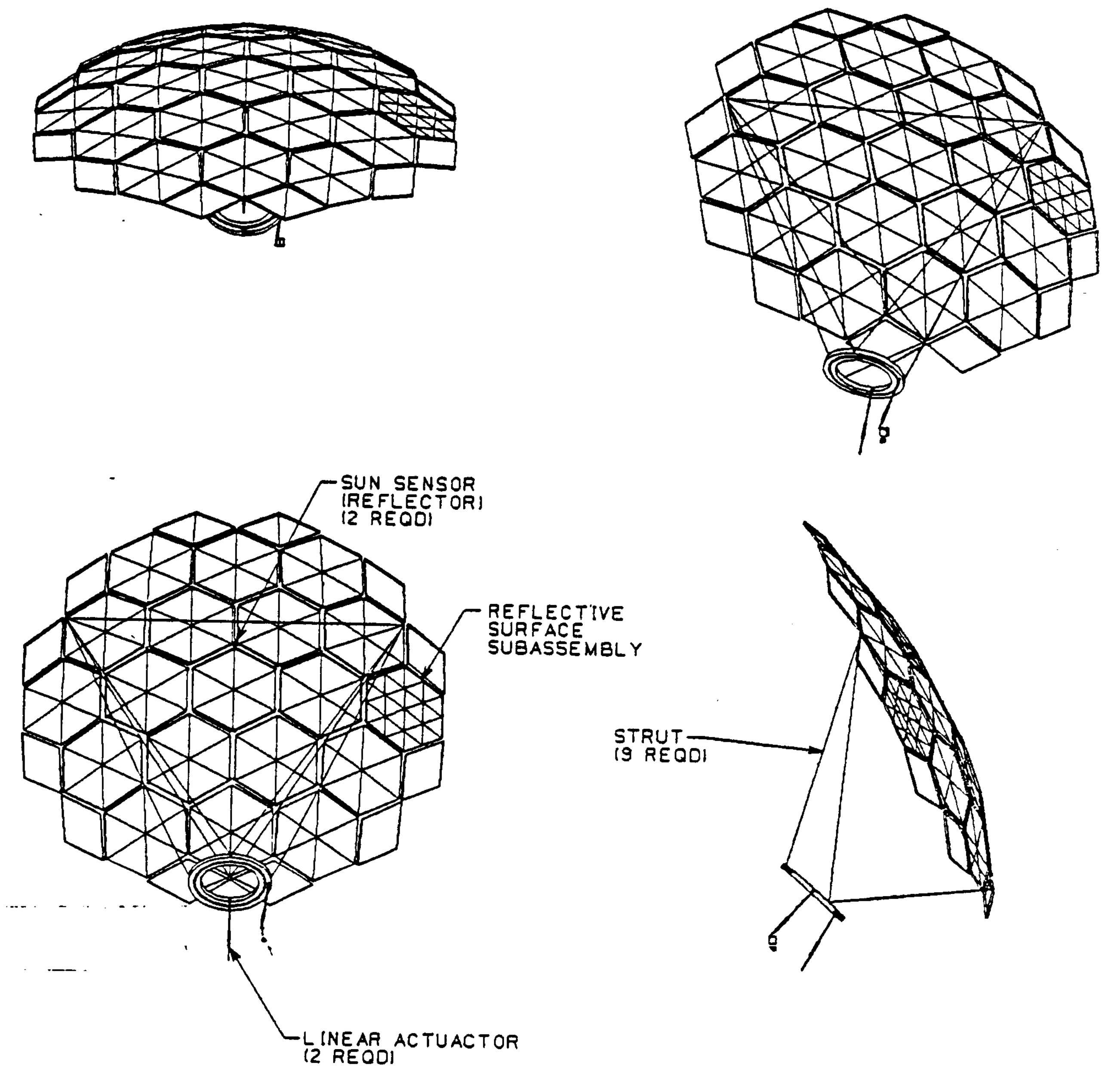


Figure 7.3-1b - Current Concentrator Model Configuration

articulate the reflector in pitch, yaw and, for some cases, focus. Total mass of the previous configuration concentrator and support structure was 941 kg (2070 lbs).

TABLE 7.3-1

PREVIOUS AND CURRENT CONCENTRATOR STRUCTURE ELEMENTS

ITEM	PREVIOUS DIMENSIONS OD (in) X WALL (in)			CURRENT DIMENSIONS OD (in) X WALL (in)		
Back-up truss	2	X	0.100	3	X	0.043
Strut	2	X	0.100	3	X	0.043
"T" Vertical	9	X	0.375	Not Applicable		
"T" Horizontal	6	X	0.250	Not Applicable		
Main mast	12	X	0.500	Not Applicable		
Gimbal Ring	Not Applicable			9 (torus)	X	0.375
Gimbal Ring Supports	Not Applicable			6		0.25
PCU Supports	Not Applicable			6		0.25

The current reflector support strut ends are modeled as fixed connections. The mass of the current configuration for the interface structure and struts has been updated in the model. The detailed design descriptions of the fine pointing mechanism, struts and interface structure are contained in sections 2.2.3 and 2.2.5 respectively.

7.3.3.1 Previous Configuration Analysis Results

Figure 7.3-2 illustrates the first flexible mode of the coupled structure for the case where the transverse boom-end of the main mast is considered cantilevered, (i.e. clamped in all six directions). The first three modes are summarized below:

First mode - .129 Hz mast torsion, some bending about the reflector diameter

Second mode - .538 Hz first reflector diametral bending

Third mode - .572 Hz reflector diametral bending orthogonal to the second mode

The reflective surface back-up truss is participating in the second and third modes; however, it was not considered a candidate for stiffening as the target mass had already been exceeded. A study was made to evaluate the

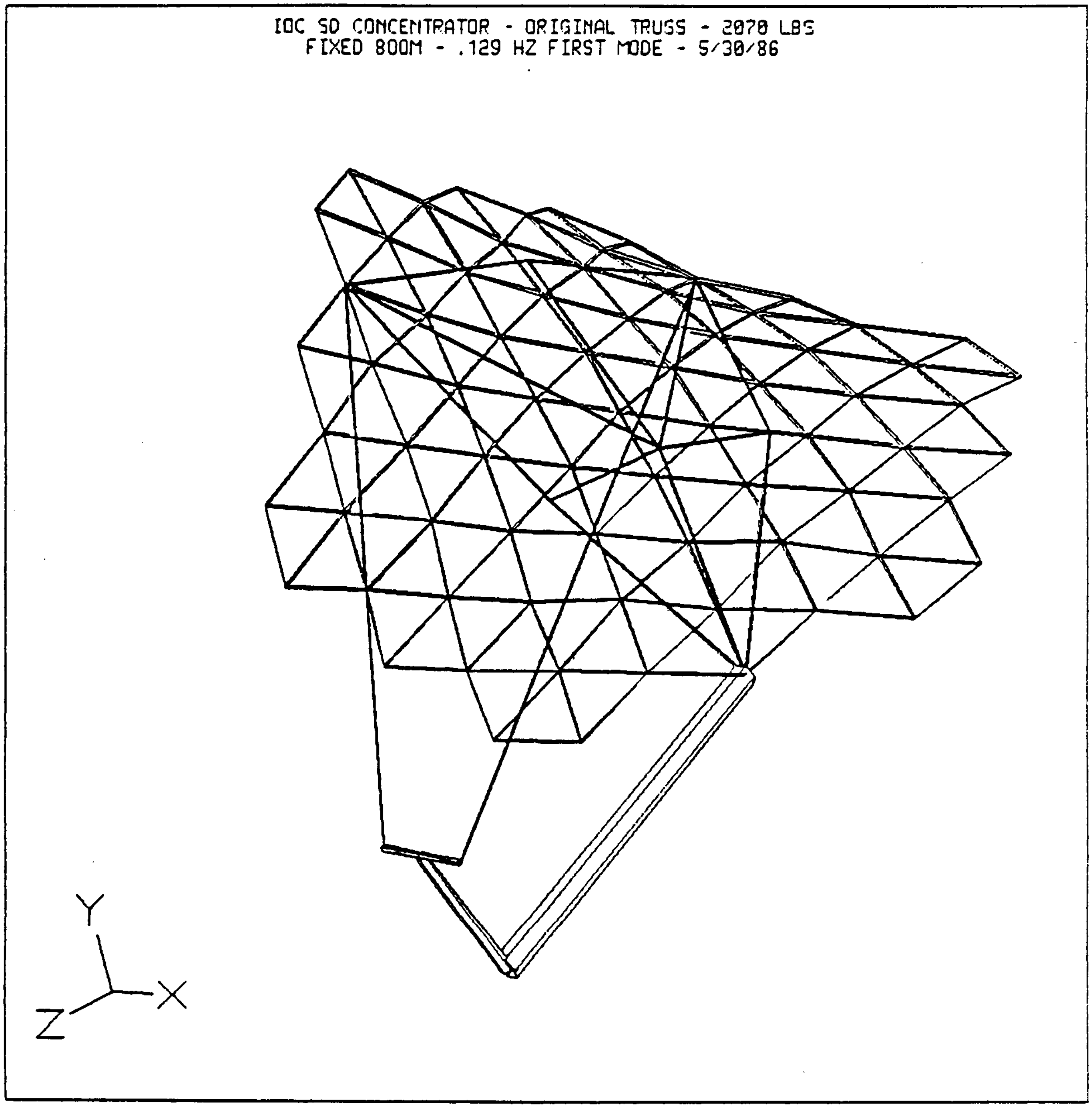


Figure 7.3-2 Previous Configuration First Mode

effects of removing the back-up truss network, which resulted in a 55 kg (120 lb) mass saving, however, the second and third modal frequencies were lowered by a factor of two from already unacceptably low values.

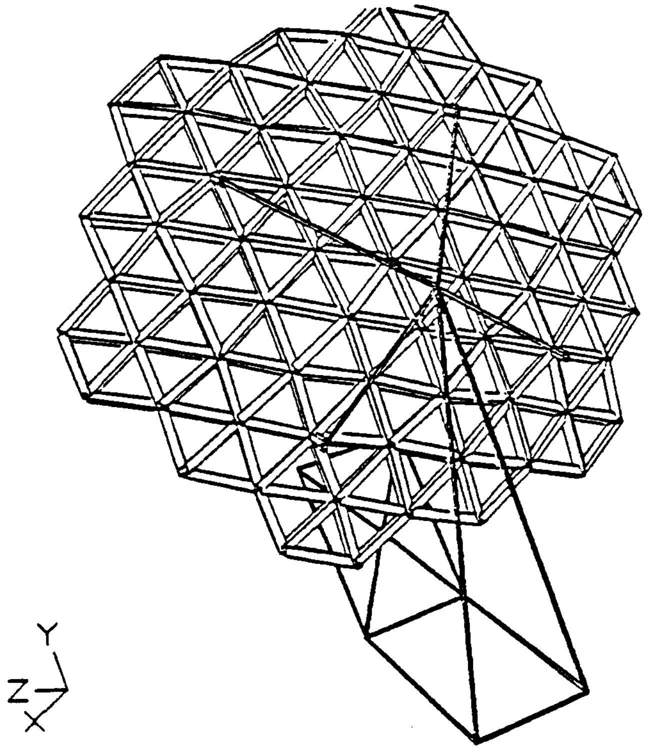
7.3.4 Previous Configuration Alternative Concepts

Figures 7.3-3 and -4 illustrate three alternative concepts used to connect and control the concentrator, all with the aim of meeting target mass and frequency characteristics. All these designs were characterized by high mass (>1045 kg) and low modal frequencies (<0.5 Hz). All designs employed four point actuation for maximum stiffness, some with rear mounts for minimum solar blockage, some with front mounts for minimum mass.

An independent evaluation of the structural dynamics of a 6-strut concentrator support configuration was completed. The design is illustrated in Figure 7.3-5. This configuration met the target mass criteria and its first mode frequency was 2.03 Hz. Consequently, it was included in the current configuration model for integrated analysis.

7.3.5 Design Optimization Using Transverse Boom-Type Construction

Using a truss network design analogous to transverse boom construction, a rear-mount four-actuator configuration was generated, as illustrated in Figure 7.3-6, and analyzed. The truss consists of pinned-pinned tubular struts of 2-in OD x .060-in wall constructed of graphite-epoxy composite. Moment release is provided in two directions at the concentrator end of each of the four actuators. Actuator connection is at the four vertices of the center hex module, thus dictating truss bay dimensions of 1.8m (6 feet) high by 3.7m (12 feet) deep. Total mass of this type of configuration, based on the previous reflector, is 773 kg (1700 lbs). For articulating the concentrator in pitch, yaw and focus, four actuators are a viable alternative to three, dual-redundant actuators in terms of fail-operational characteristics. Trade-offs exist with regard to mass and cost penalties associated with the use of three dual redundant actuators versus four non-redundant actuators, each with a declutching capability. Optimization of the control loop is addressed in Section 7.4.



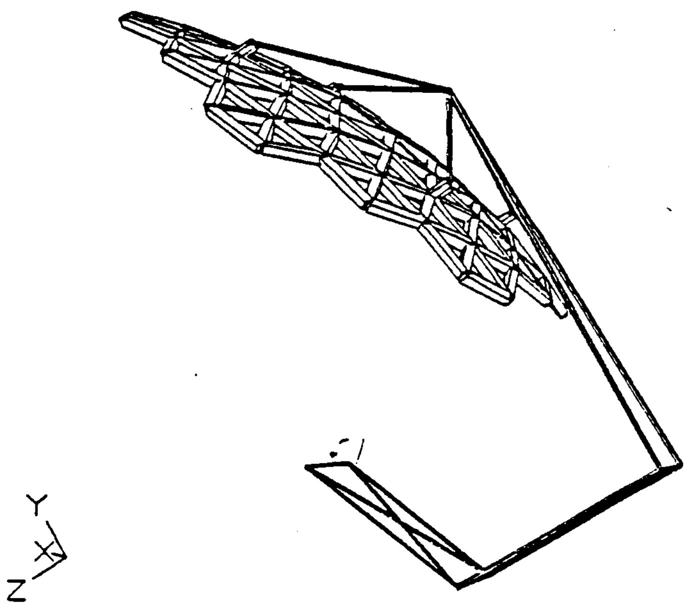


Figure 7.3-3 Alternate Concepts

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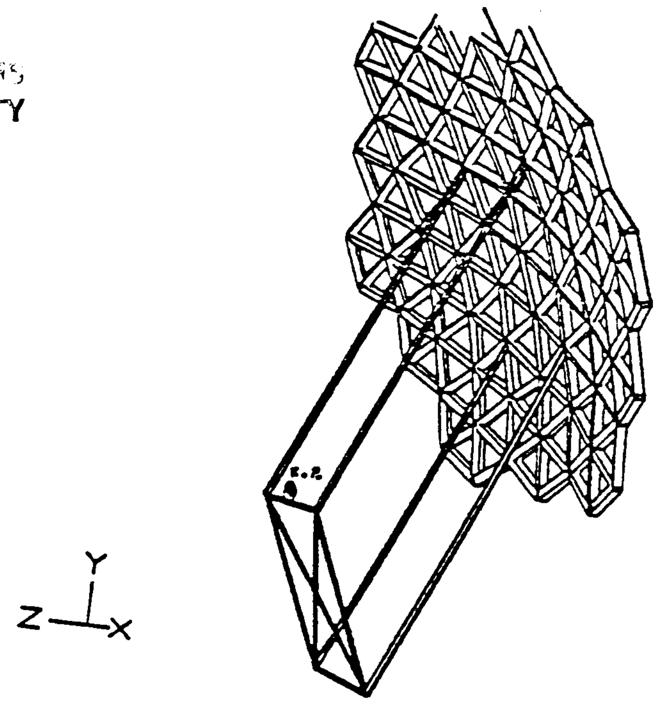

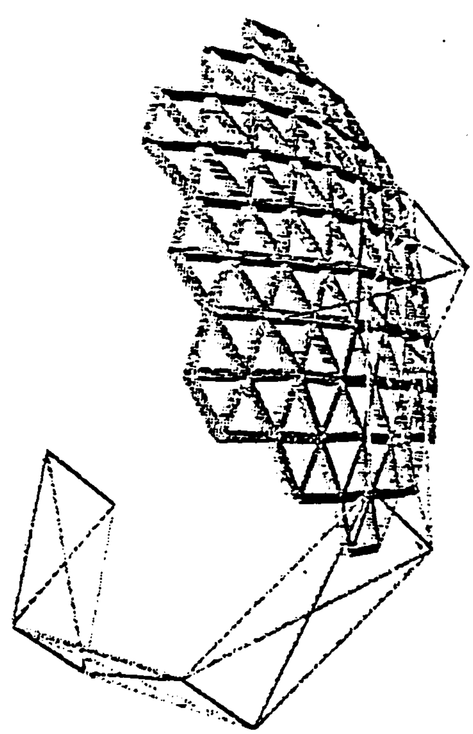

Figure 7.3-4 Alternate Concepts (continued)

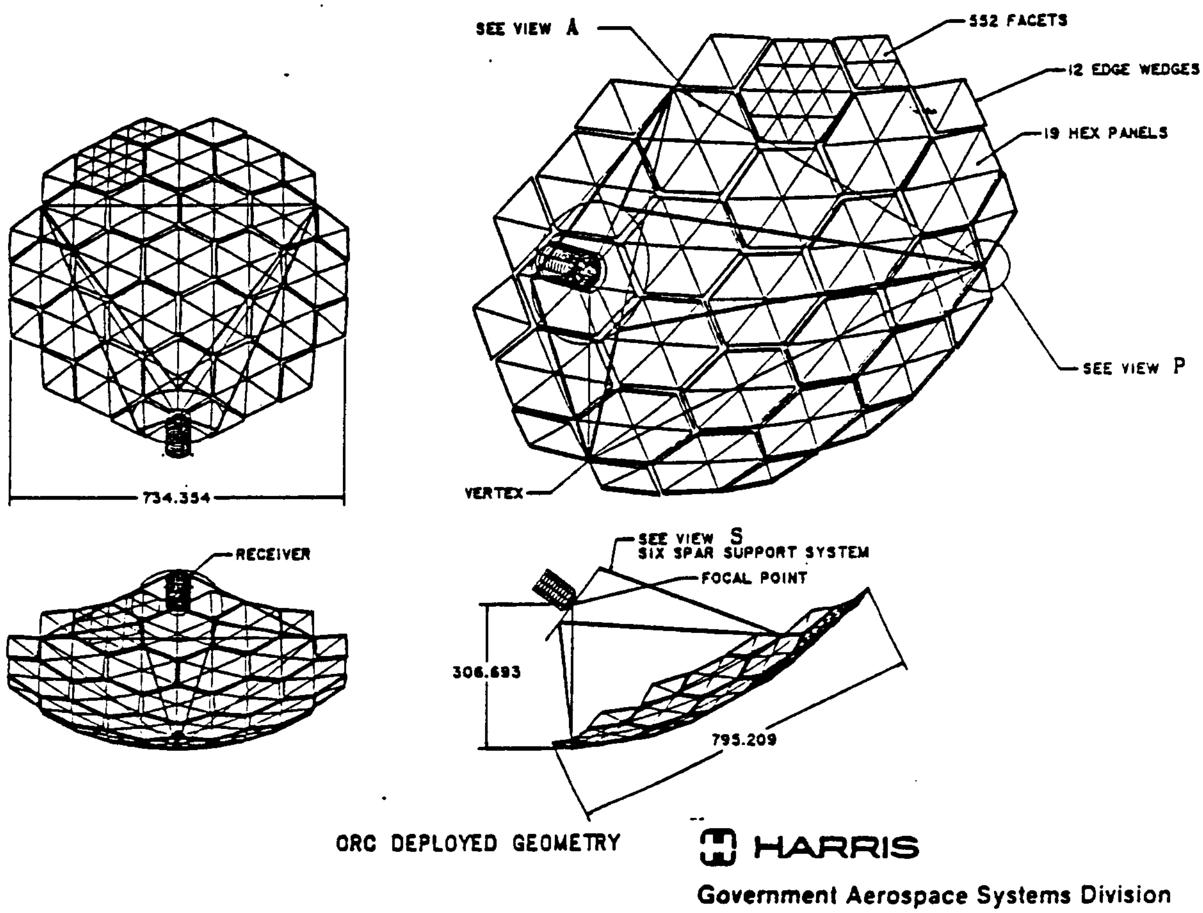

Figure 7.3-5 Six Strut Reflector Structure Configuration

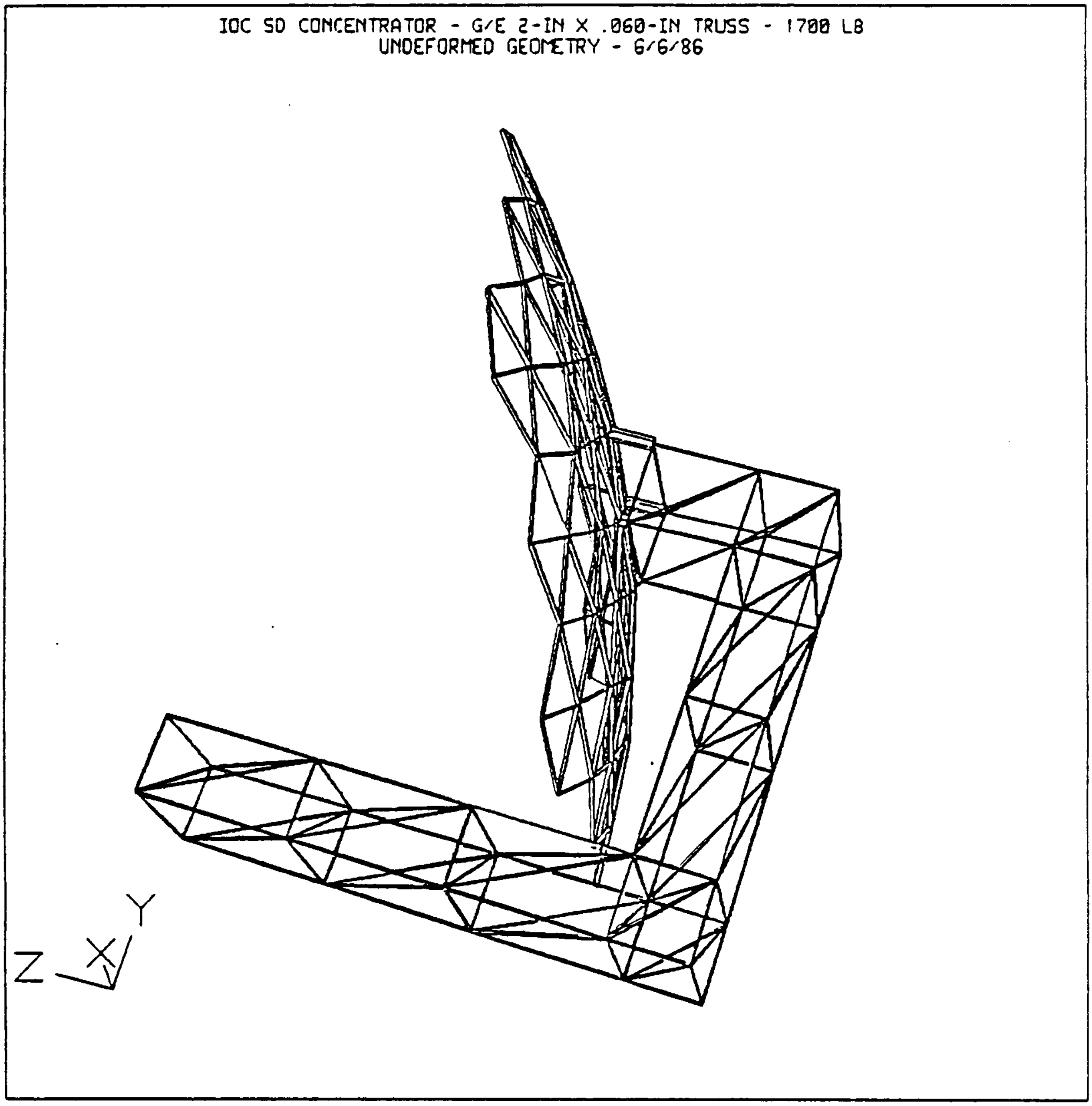

Figure 7.3-6 Back Mounted Truss Configuration

7.3.5.1 Truss Concentrator Support Modal Analysis

Vibrational modes derived for the design illustrated in Figure 7.3-6 include a 1.089 Hz first mode consisting of reflector diametral bending, shown in Figure 7.3-7, and a 1.255 Hz second mode, consisting of bending of the actuators and some fore-aft bending of the upright interface structure. The second mode could benefit from structural stiffening at the actuator attach points.

7.3.5.2 Truss Design Optimization

By way of design iteration, the configuration illustrated in Figure 7.3-6 was modified such that the truss top and side sway braces were changed from being parallel to the orientation depicted in Figure 7.3-8. Although no change in structure mass resulted, this simple change yielded a 3% increase in the first modal frequency. Recent concerns about the availability of EVA resources, and their priority of application during concentrator assembly, have resulted in a de-emphasis of the truss type reflector support concept.

7.3.6 Three-Actuator Truss Design

In support of the three-versus-four actuator trade study, the interface structure configuration illustrated in Figure 7.3-8 was modified to permit the use of three fine pointing control actuators, instead of four. This new configuration is illustrated in Figure 7.3-9. No net weight savings was realized in this case, as structural stiffening at the actuator base was necessary, thus yielding the 773 kg (1700 lb) structure mass as before. Net mass savings would probably be realized in final optimized detail design of this configuration.

7.3.6.1 Three-Actuator Configuration Modal Analysis

The first mode calculated for the three-actuator configuration with mount stiffening is provided in Figure 7.3-10. As in all cases, the transverse boom end of the support truss is considered cantilevered. As seen from Figure 7.3-10, the first system mode at 1.033 Hz involves only participation by the reflector. The 1.158 Hz second mode and 1.231 Hz third mode are the result of

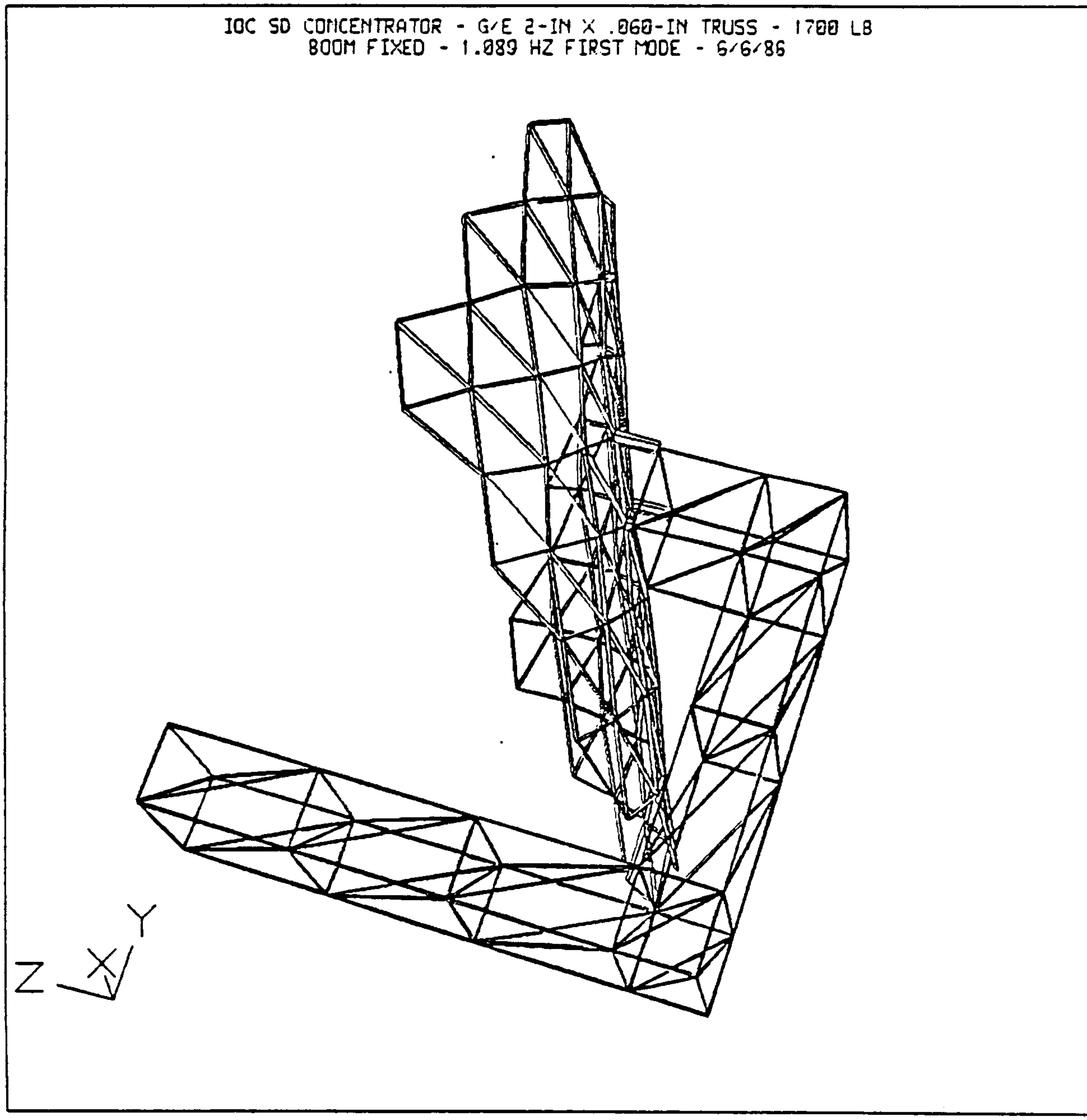

Figure 7.3-7 Back Mounted Truss Configuration First Mode

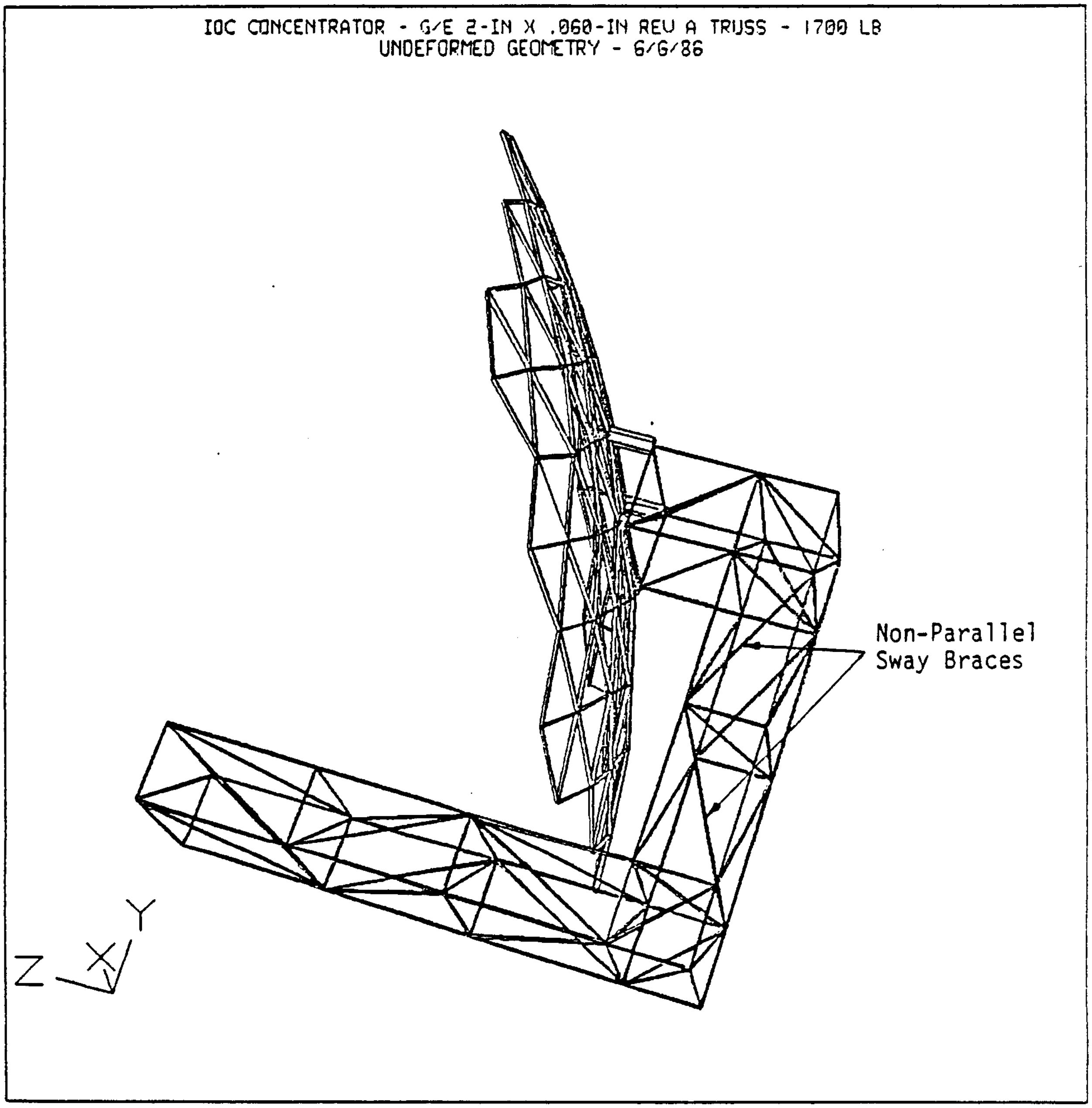

Figure 7.3-8 Modified Back Mounted Truss Configuration

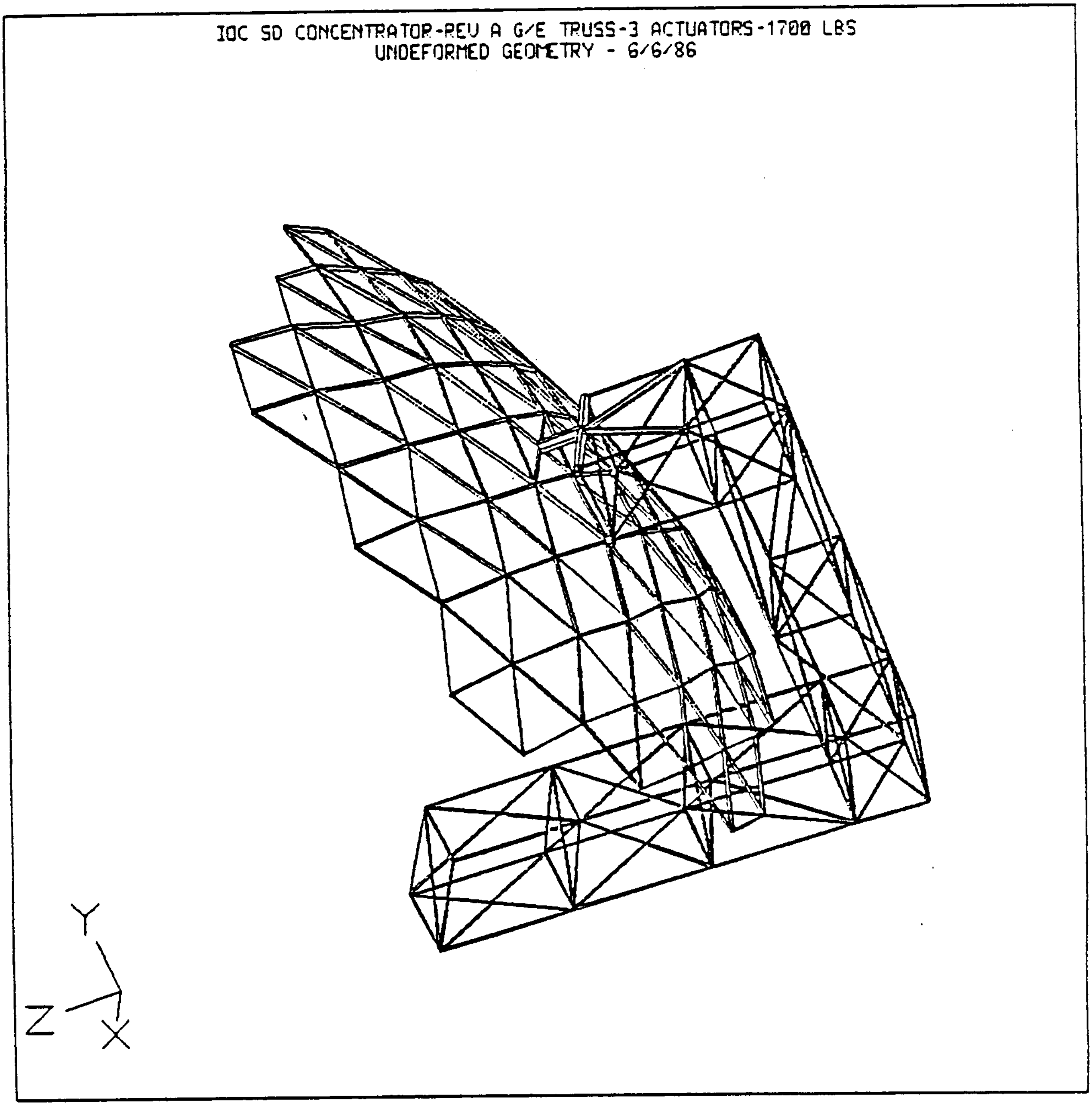

Figure 7.3-9 Three Actuator Back Mounted Truss Configuration

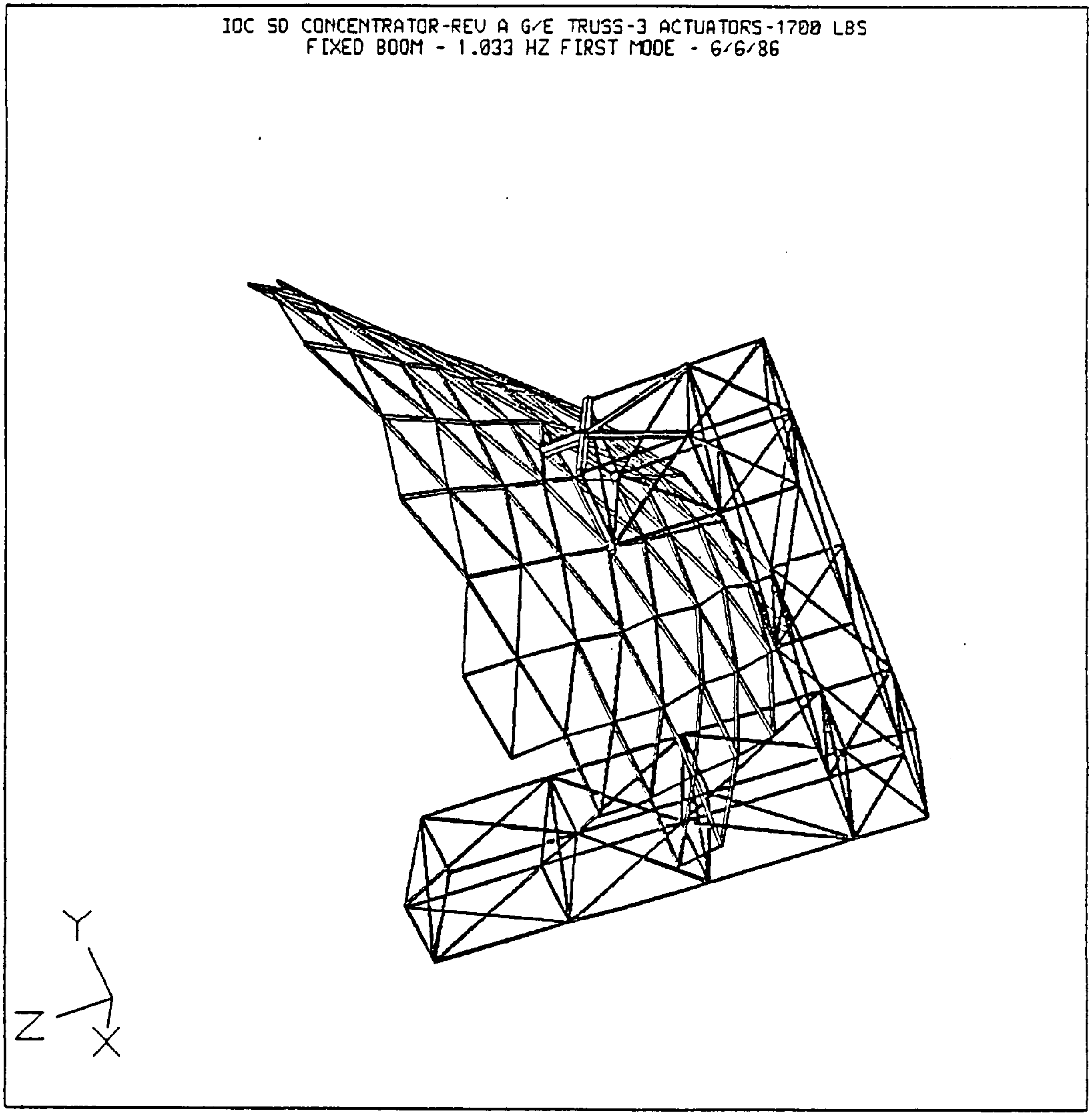

Figure 7.3-10 Back Mounted Three Actuator Configuration First Mode

coupling between the reflector and support structure. Thus, while the second and third modes would benefit from a more optimized truss design, the critical first mode is not significantly affected by the truss structure configuration. Based on the results of this investigation, it can be concluded that target mass and structural frequency constraints can be met with three fine-pointing control actuators. However, additional modal frequency margin can be obtained using four fine pointing control actuators.

7.3.7 Front Mounting Interface Structure

By way of design iteration, a candidate interface truss structure was conceived wherein mounting provisions are made from the reflective surface side of the concentrator, rather than the rear side. This configuration benefits from reduced mass, due to the shorter reach to the receiver aperture. Figure 7.3-11 illustrates one such candidate design. It is noted that the significant negative impact of such front mounting on solar energy intercept due to truss shadowing was not quantitatively assessed.

Mounting provisions are made for three actuators with mount stiffening as before. To minimize mass, attachment beams have been installed at the lateral vertices of the center hex module,for this concept, thus yielding truss bay dimensions of 1.8m (6 ft) by 1.8m (6 ft). Mount beams on the center hex consist of 10cm (4-in) x .4mm (.150-in) filament wound graphite-epoxy tubes. Total mass for this configuration is 677 kg (1490 lbs).

7.3.7.1 Front Mounted Truss Modal Analysis

The first mode calculated for this design is illustrated in Figure 7.3-12. Here again, the lower modes can be characterized as being dominated by reflector modes, with limited participation by the support structure. The 1.063 Hz first mode and 1.276 Hz second mode are mainly concentrator diametral modes with vertical and lateral node lines, respectively. The 1.359 Hz third mode can be characterized as vertical bending of the fine-pointing actuators and support mount with vertical translation of the concentrator. Some improvement in the modal frequency of this mode can be achieved with optimized actuator mount design. Due to the essentially equivalent performance with equal support structure mass and decreased reflector shadowing of the current

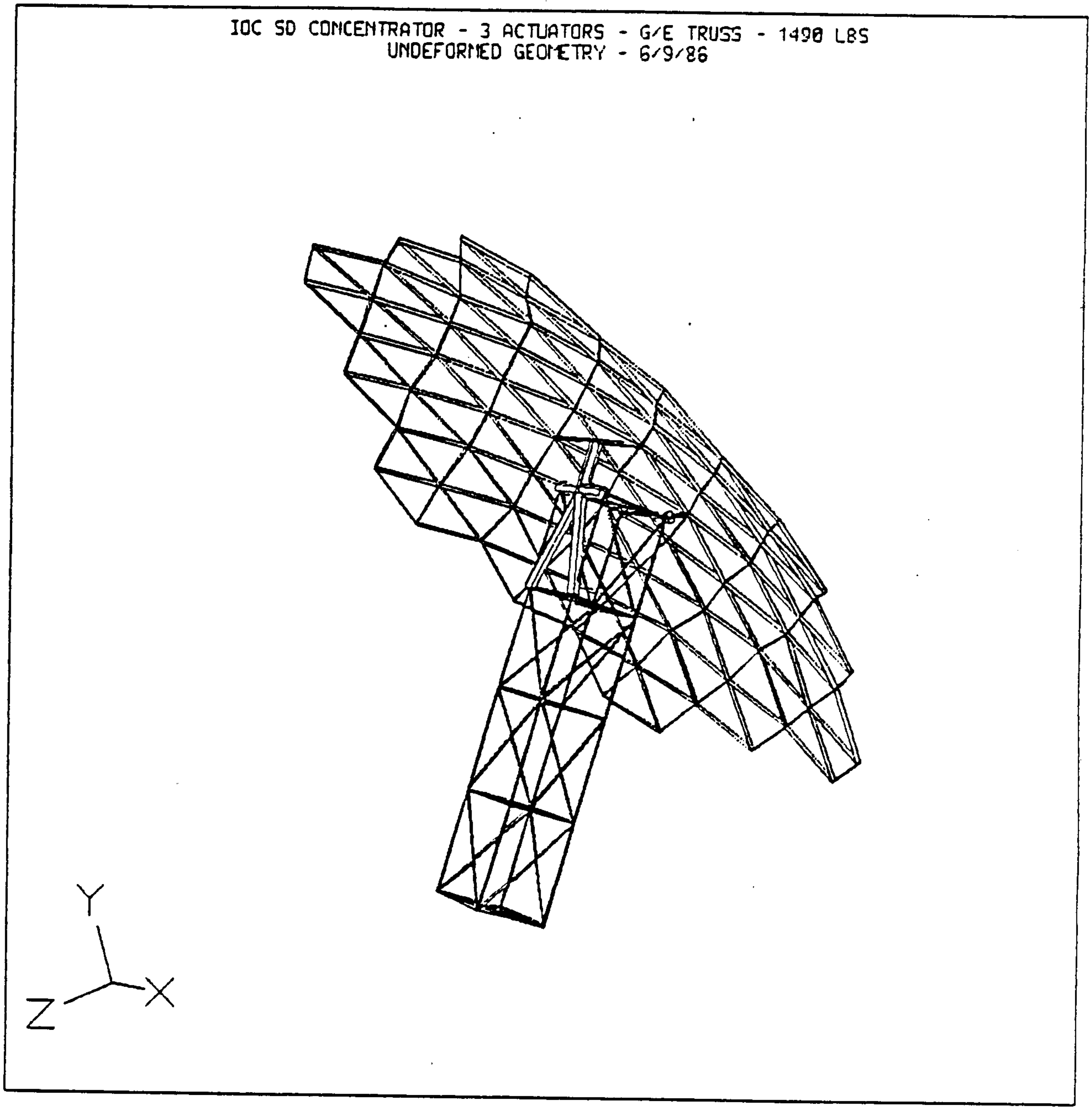

Figure 7.3-11 Front Mounted Truss Configuration

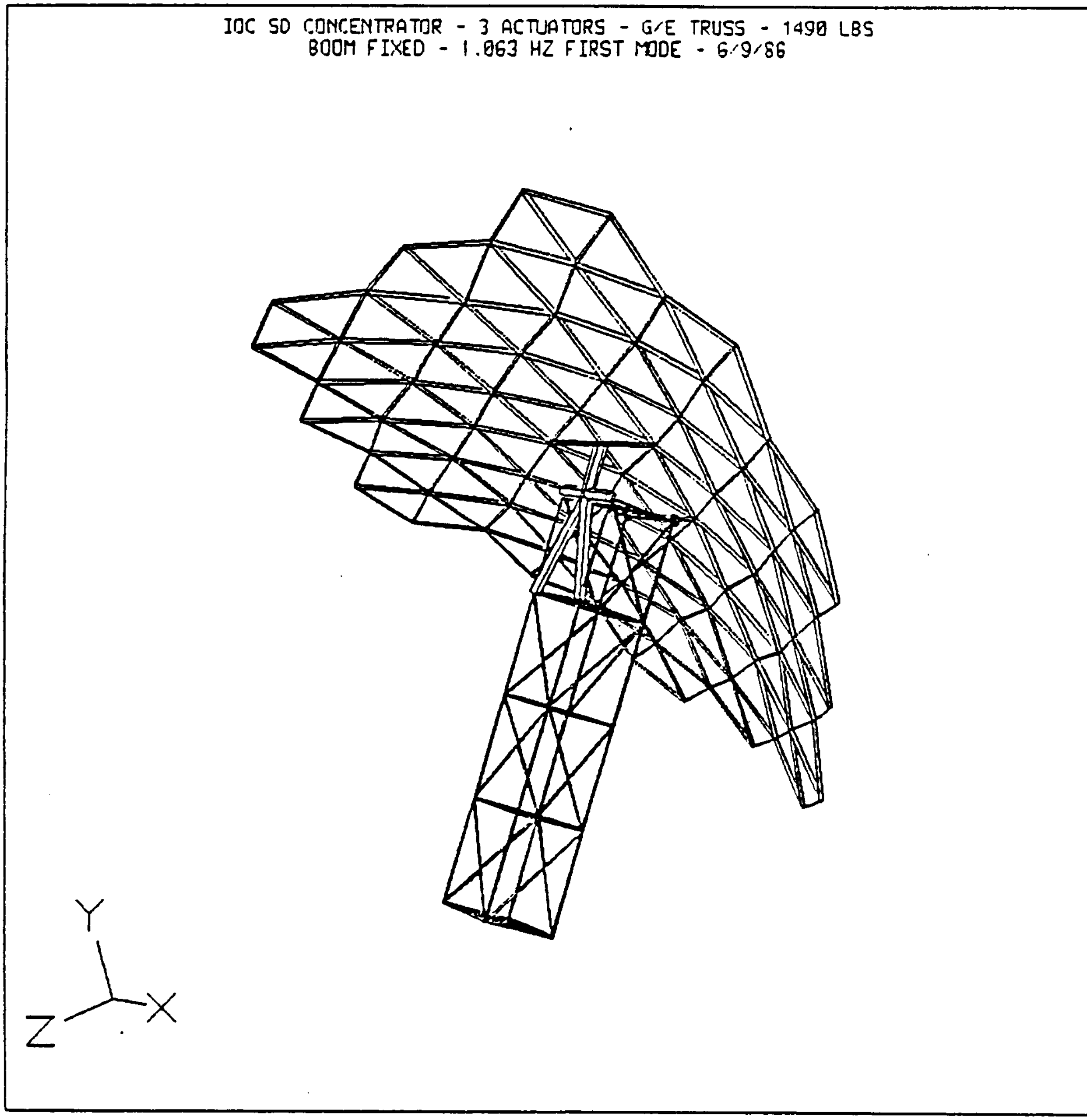

Figure 7.3-12 Front Mounted Truss Configuration First Mode

configuration, the front mounted truss configuration was not selected.

7.3.8 Current Configuration

The current configuration, shown in Figure 7.3-13, was synthesized as a result of structural dynamics and optical performance (see section 7.4) drivers. This configuration achieves the fine pointing function through the use of a double ring-gimbal mechanism supported by an A frame attached to the interface structure. The linear actuators are currently located between the fine pointing mechanism and the interface structure. this configuration has a mass penalty associated with the fine pointing mechanism (which has not yet been mass optimized) but allows an engineered solution to the problem of concentrator stiffness through independent stiffening of the gimbal rings. The dimensions of the structural elements of the current configuration are listed in section 7.3.3.

7.3.8.1 Current Configuration Analysis

The current configuration first mode is illustrated in Figure 7.3-14. IN this case the lower mode, 0.976 Hz., which is the sixteenth transverse boom mode, is characterized by gimbal ring bending out of plane, with a significant contribution form the beta joint, resulting in a concentrator rigid body sway. The second and third modes are 1.37 Hz. and 2.09 Hz respectively. Some improvement in stiffness to weight of this configuration can be achieved by optimizing the gimbal ring design. Based on the currently available improvement in structural and optical performance offered by this concept (see section 7.4) over the previous concept and its derivatives, this concept was selected and is recommended as the final preliminary design reference configuration for the concentrator structure.

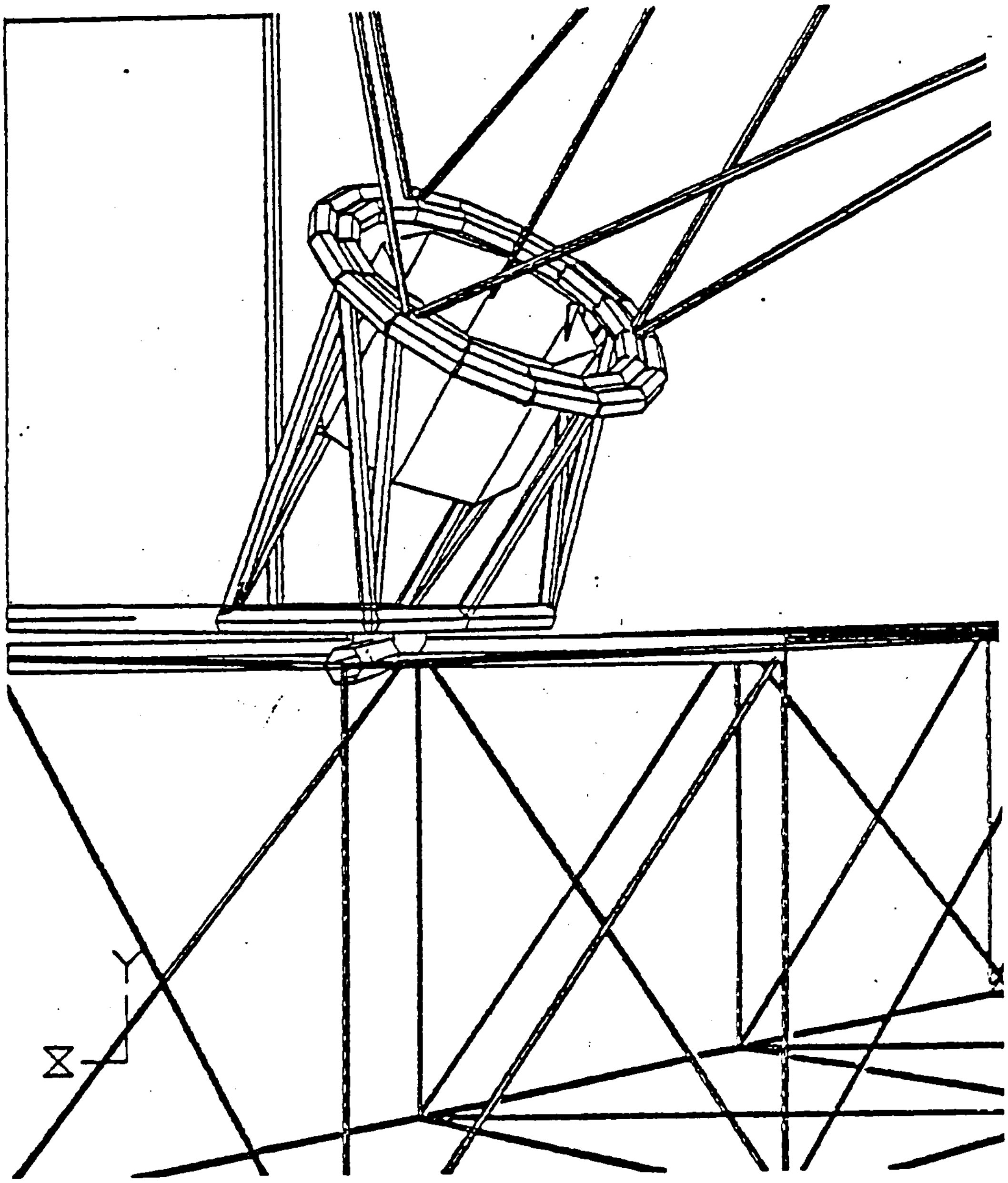

Figure 7.3-13 Current Configuration

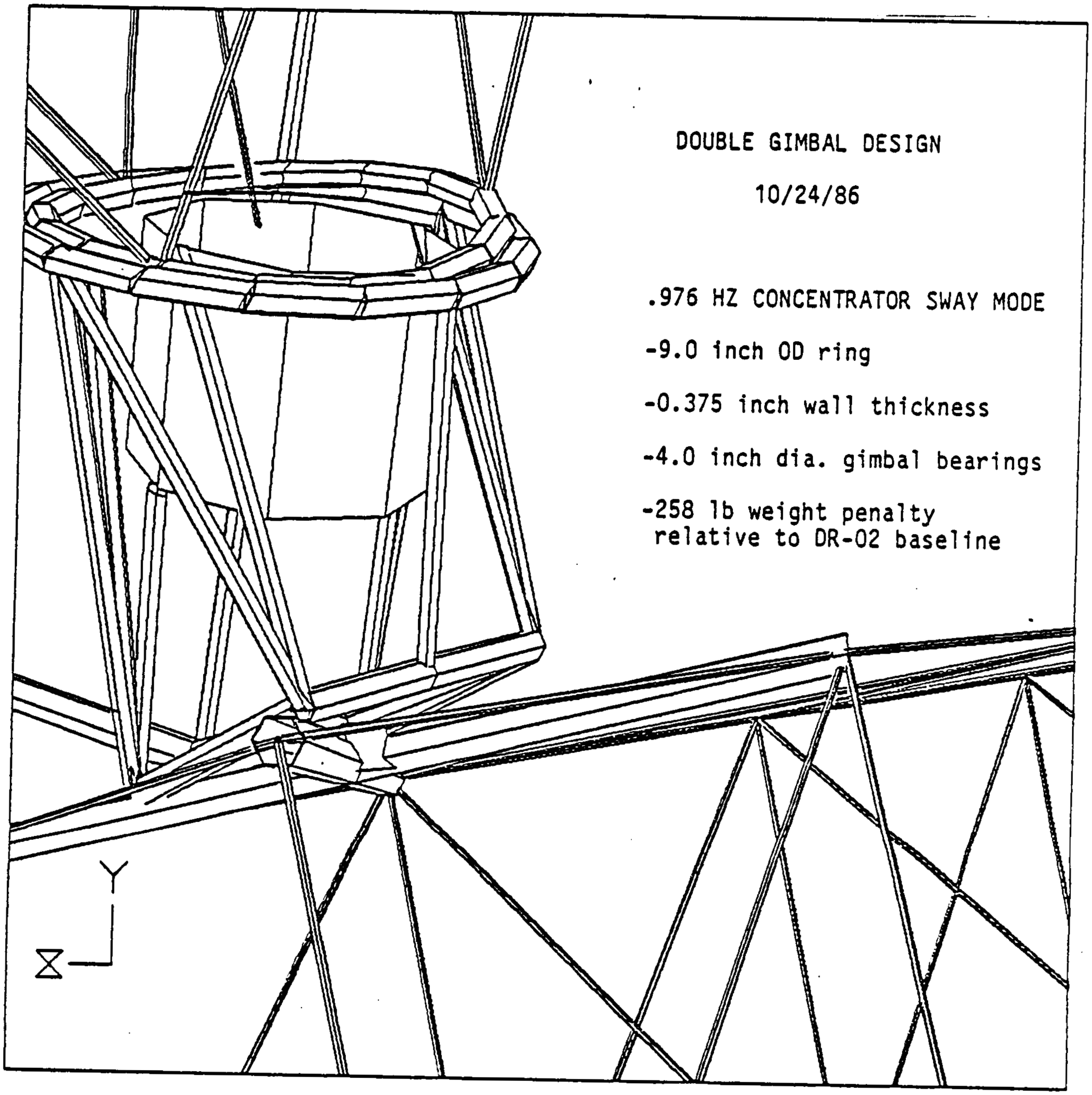

Figure 7.3-14 Current Configuration First Mode

7.3.9 Conclusions

It was concluded from the results of this study that the mass constraints and modal frequency requirements for the coupled reflector/structure system are achievable using the current configuration. It has similarly been shown that four fine-pointing control actuators provide a reasonable amount of margin in the fundamental frequency. It is expected that issues of failure modes and effects, cost, and maintenance outweigh the structural benefits of the four actuators. Lastly, a variety of issues not in the scope of this study need to be addressed with regard to design optimization of the current structure and fine pointing control methodology. These include, EVA time for assembly, STS packaging commonality with other Space Station structural and assembly elements, levels of tolerable degraded power to the user, the effects of structure joint deadband, stiction in multi-jointed structures, and redundancy management, if any, for the fine-pointing control loop.

7.4 CONCENTRATOR CONTROL OPTIONS

7.4.1 Summary

A fine-pointing concentrator control option evaluation was completed in support of the concentrator preliminary design. The objective of the study was to evaluate several fine-pointing control concepts in terms of control loop logic and suitability for this application. The optical performance of four of these concepts was also evaluated. The evaluation criteria included control simplicity, authority and error budgets, as well as optical performance. The specific objective was to identify the concentrator control requirements and to specify concentrator control performance. The control loop bandwidth was selected at 0.5 Hz, based on structural dynamics and Space Station controller bandwidths inboard of the SD subsystem. It was concluded, as a result of this study, that viable control loops for concentrator fine-pointing control can be of a simple variety and that the optical performance of the reference configuration is acceptable, based on the data obtained to date.

7.4.2 Strut/Universal Joint Configurations and Options

In previous issues of DR02, the reference concept for fine-pointing actuation employed a steerable reflector oriented by shortening or lengthening two of three struts connecting the concentrator to the interface structure, as shown in Figure 7.4-1. Five 2-axis universal joints were used to avoid bending moments in the struts or reflector structure. With length-positioning actuators on two of the struts, the concentrator may be pitched or tilted about the parabola vertex with respect to the interface structure which is not shown in the Figure. Since the receiver is rigidly attached to the interface structure, it is possible to orient the reflector continuously so that its concentrated image is centered in the receiver.

This optical system, however, presents three issues: (1) At any off-axis sun angle, the focus will be displaced with respect to the receiver aperture; (2) At an off-axis angle, the focal spot will expand and distort; (3) The structural stiffness of the system is primarily limited by the stiffness of the mast strut. Each of these three issues is discussed elsewhere in this section.

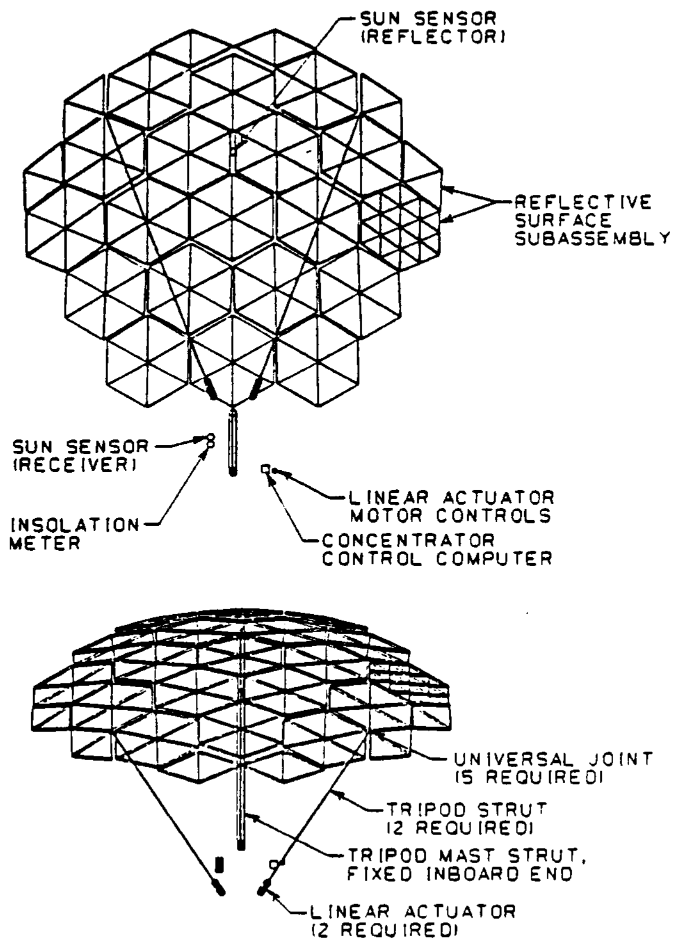

Figure 7.4-1 Previous Reference Fine Pointing Concept

In response to these issues, three major configuration options were evaluated : (1) An integrated system of the type presented by LeRC on April 16-17, 1986 (Figure 7.4-2); and (2) a variation on the previous reference concept with respect to strut, universal joint, and actuator arrangements, illustrated in Figure 7.4-3; and the concept which was ultimately selected as the current reference, shown in Figure 7.4-4, wherein the fine pointing mechanism consists of a dual ring-gimbal and two linear actuators mounted on the interface structure.

The integrated concept, Figure 7.4-2, avoided all three issues cited above for the previous reference concept. In trade, however, the greater inertia of the integrated concept required increased actuator loads and power, flexing of fluid lines, and perhaps, increased stiffness of the transverse boom.

Nevertheless, the integrated concept remains an attractive option and is a strong alternative to the current reference concentrator fine pointing configuration.

The second candidate was a variation on the previous reference concept. This concept, herein termed the "nominal- alternative" concept is shown in Figure 7.4-3. Because its struts are fixed at their bases, all participate in structural resistance to disturbances about the optical axis-- which, as discussed in Section 7.3, was the weakest mode for the previous reference concept. In addition, because all struts are actuated, a second deficiency of the previous reference concept is resolved; i.e., the focus can be positioned along the receiver optical axis. However, this concept can have, under some circumstances, a smeared and distorted focus which affects the solar intercept factor.

The third candidate, shown in Figure 7.4-4, features a two axis fine pointing mechanism which gimbals the reflector independently of the PCU, resulting in a low gimbaled mass and modest coarse and fine pointing parasitic power requirements. The two-axis fine pointing mechanism kinematically constrains the reflector focal point and effectively eliminates translation of the focal point with respect to the receiver aperture, resulting in a simplified optical system. However, this candidate has a greater fine pointing inertia than the

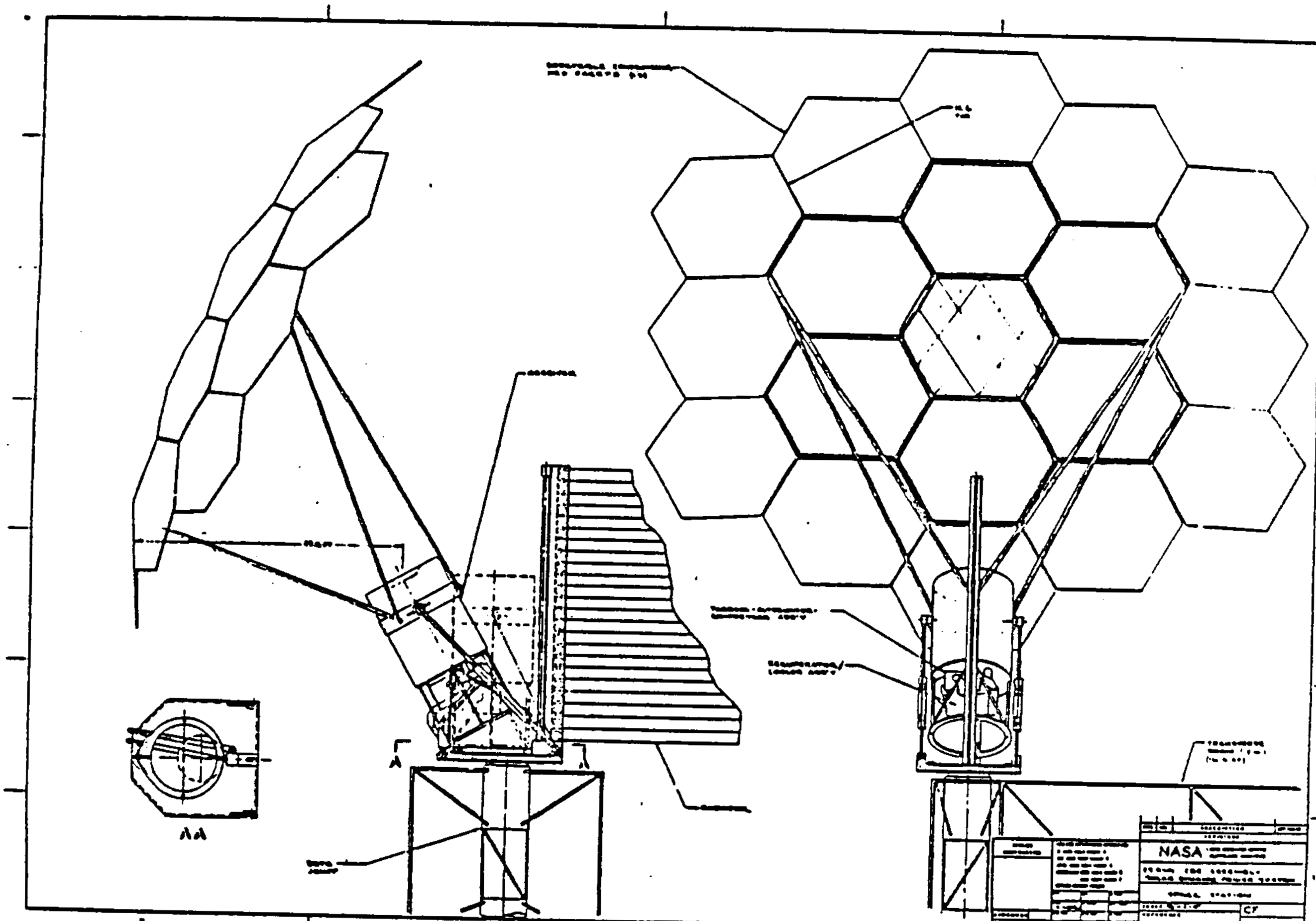

Figure 7.4-2 Integrated Alternate Concept

FINE POINTING SYSTEM CONCEPT

(3 ACTUATORS, 2 AXIS TILT PLUS FOCUS)

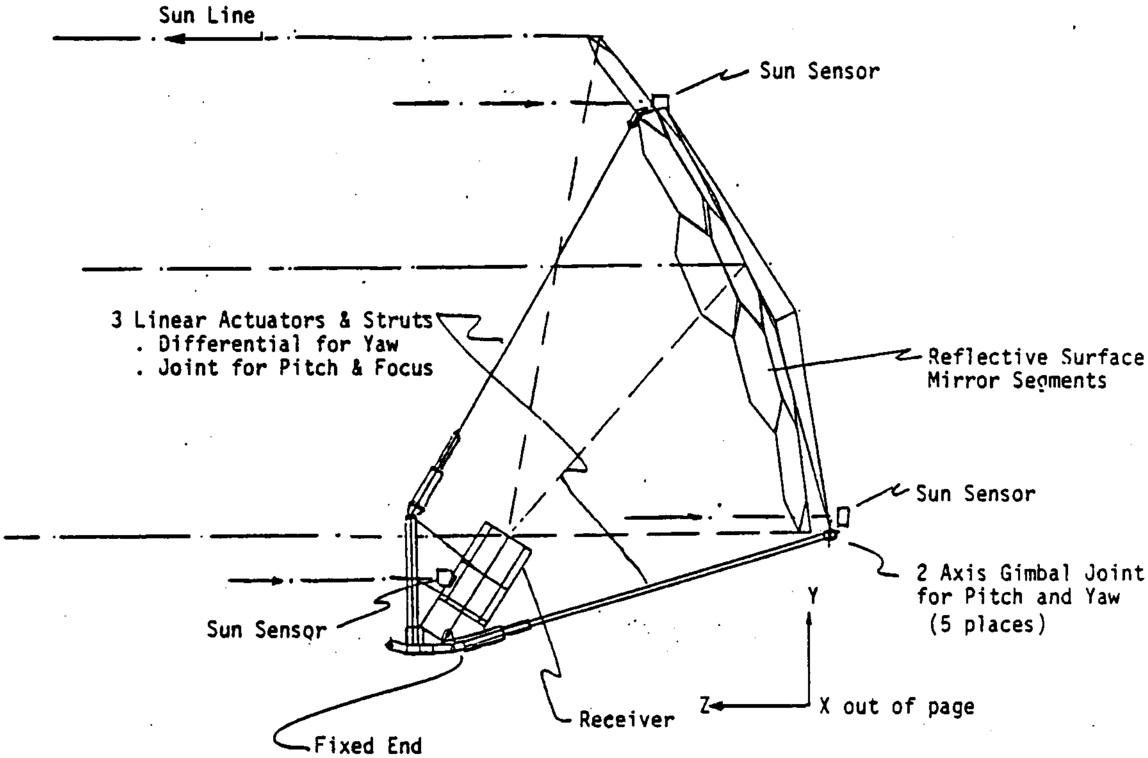

Figure 7.4-3

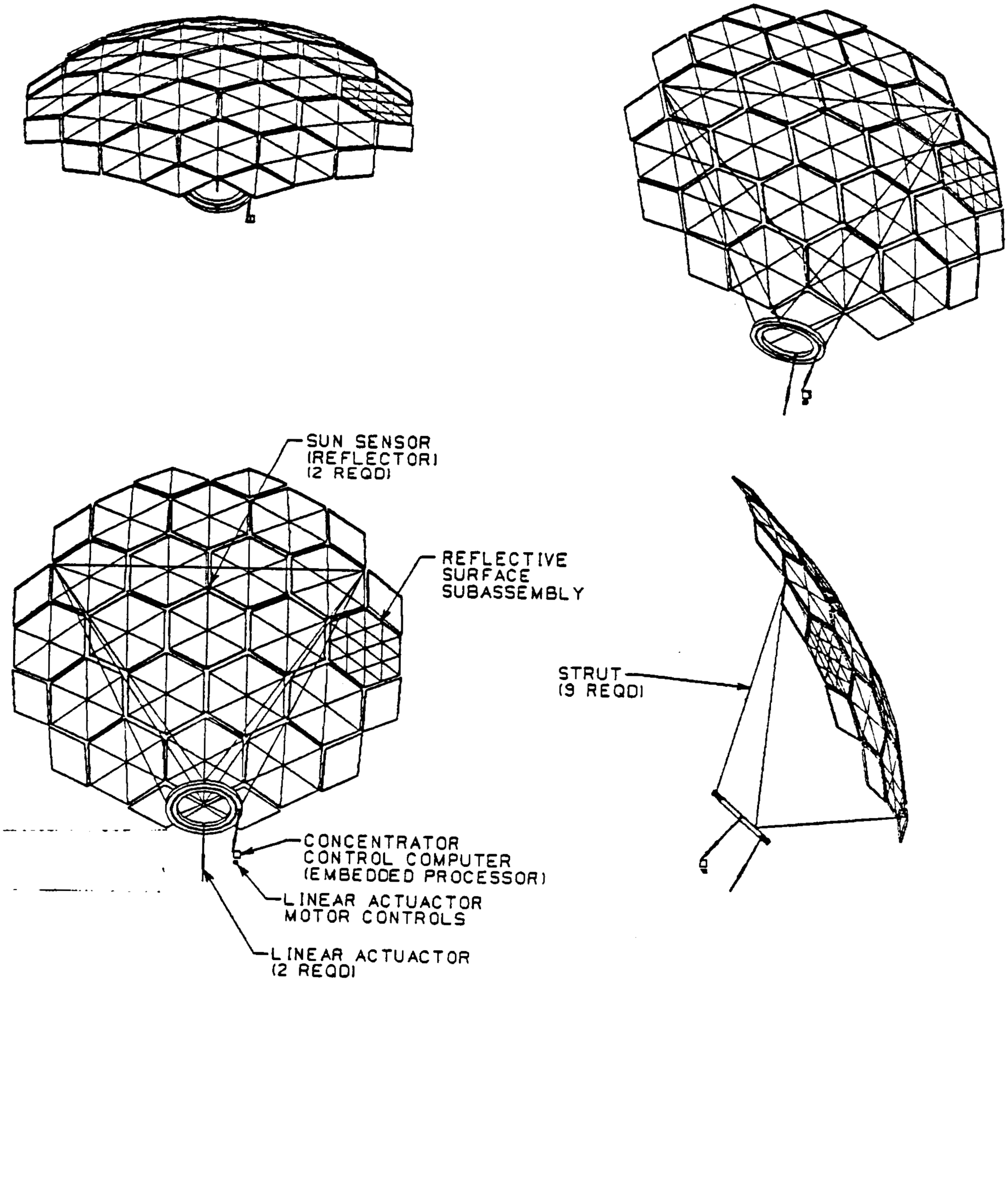

Figure 7.4-4 Current Reference Concept

previous reference concept. The inertia about an axis parallel to the alpha axis is 150% of that of the previous reference concept. The inertia about an axis parallel to the beta axis is about 3 times that of the previous reference concept. However, the inertia of this concept is significantly lower than that of the integrated concept. The increased inertia of the third candidate was judged to be acceptable in terms of the linear actuator forces required. This candidate also exhibits a 300 lbm penalty with respect to the previous reference concept. In spite of these detrimental features the third candidate was selected on the basis of superior optical performance, structural stiffness and control simplicity.

7.4.3 Fine-Pointing Control

7.4.3.1 Requirements

The objective of the Fine Pointing Control Loop (FPCL) is to maximize the amount of solar energy intercepted by the receiver. Prior studies (e.g., DR-19, DP 4.2) have shown that the overall solar dynamic power subsystem is optimized when the FPCL is capable of pointing within 0.1 degree for the CBC SD subsystem, which has the most stringent pointing requirement of the two cycles under consideration.

This requirement, then, has been levied on the FPCL and is defined as follows:

> The centroid of the "Circle of Least Confusion" (CLC)* shall lie within a radial displacement of ϵ_r , the center of the receiver aperture--as measured within the plane of the CLC. The time average of ϵ_θ shall be equal to or less than 0.1 degrees during the sunlit phase of any orbital period--with an estimated confidence of at least 99.7%. ϵ_r and ϵ_θ are defined in Figure 7.4-5.

The FPCL shall satisfy the above requirement within the environment created by the Space Station (SS) orbital state, control system, and operations. This environment is briefly outlined below.

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* The CLC is described in Fundamentals of Optics, Jenkins and White, McGraw Hill, 1957

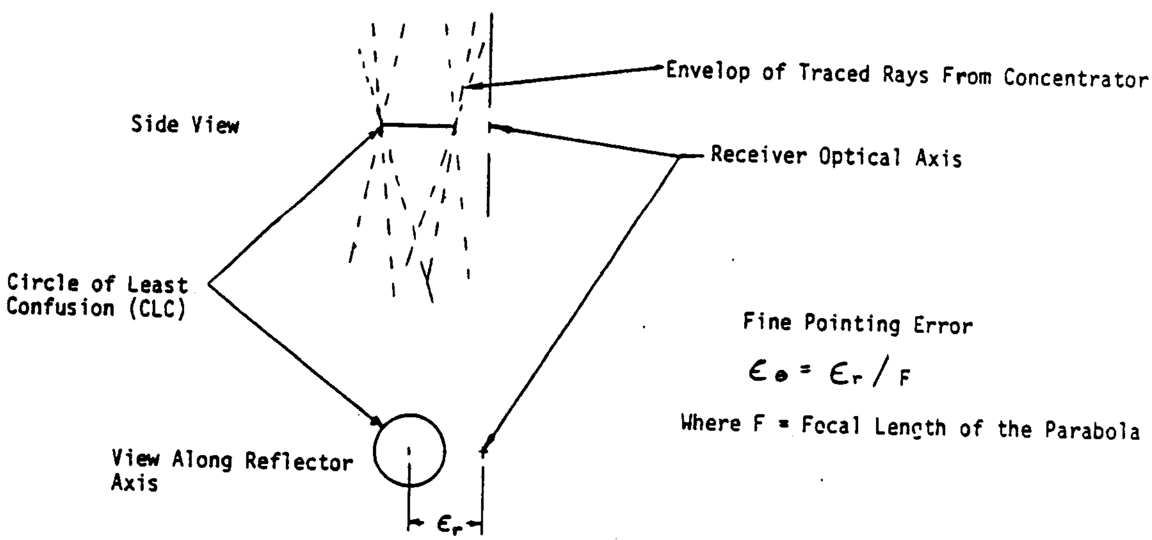

Figure 7.4-5 Definition of Fine Pointing Error Terms

7.4.3.2 Environment

1. Motions and distortions of the solar dynamic power module induced by the orbital state.
   1.1 Solar/thermal-induced distortion of the concentrator structure; i.e., eclipse transients and shadowing. This is a long-period effect which can cause uncertainty regarding the intercept factor.
   1.2 Aerodynamic drag-induced distortion of the concentrator structure/figure.

2. Gimbal control of the alpha/beta joints.
   2.1 Long-period (<< 0.01 Hz) misalignments of the alpha/beta joints due to initial installation, thermal expansion, and/or mechanical creep in the boom structure. This effect creates an offset which must be corrected by the FPCL and, therefore, drives the dynamic range of the system.

   2.2 Short-period (>> 0.01 Hz) dynamic motion of the alpha/beta joints due to joint non-linearity and control system cycling. These dynamic motions will result from alpha/beta joint control lags, joint backlash, and flex modes of the transverse boom as affected by SS attitude control and operations.

3. Transients induced by aperiodic events such as orbiter berthing, manipulator operations, and SS propulsion (i.e., orbit make-up and CMG desaturation).

These environments are, of course, subject to uncertainty at this time--but reasonably bounded values may be derived for the most serious factors affecting the FPCL performance.

7.4.3.3 Control Loop Concept

Figure 7.4-6 characterizes one of the major control issues affecting the design of the FPCL. The issues are:

1. The fine-pointing control bandwidth must be high enough to preclude interaction with the control modes of the alpha/beta joints, and low enough to avoid structural interaction with the concentrator and support structure.

2. There must be sufficient gain to achieve the required pointing accuracy in the presence of the disturbance environments.

Data from Work Package 2 indicates that the control frequency for the alpha/beta joints will be about 0.04 Hz. Finite element evaluations (reported in Section 7.3) indicate that a light-weight concentrator structure with at least a 1 Hz fundamental mode is achievable. Therefore, a window of nearly 1-1/2 decades exists for setting the control frequency. On this basis, the loop gain has been tentatively set to achieve cross-over at approximately 0.2 Hz.

The fine-pointing control concept is schematically illustrated in Figure 7.4-7. Multiple sensors are rigidly attached to the reflector at a location near the strut/reflector joint. The averaged sun vector is therefore derived relative to the plane formed by the strut-concentrator junctions. The position of this plane relative to the receiver can be derived from knowledge of the strut lengths and the receiver position referenced to the lower ends of the struts. These data are available from fabrication and initial alignment procedures. (A following discussion of error budgeting will address the uncertainties in these data.) The controller processor effects this determination and also processes position and rate data to achieve stability margins. In addition, this processor determines the length adjustments required for each of the two linear actuators

Since the receiver is rigidly (for all practical purposes) attached to the interface structure, the sensing of a sun vector error directly translates into a sensed alpha and beta displacement which must be corrected by rotating the

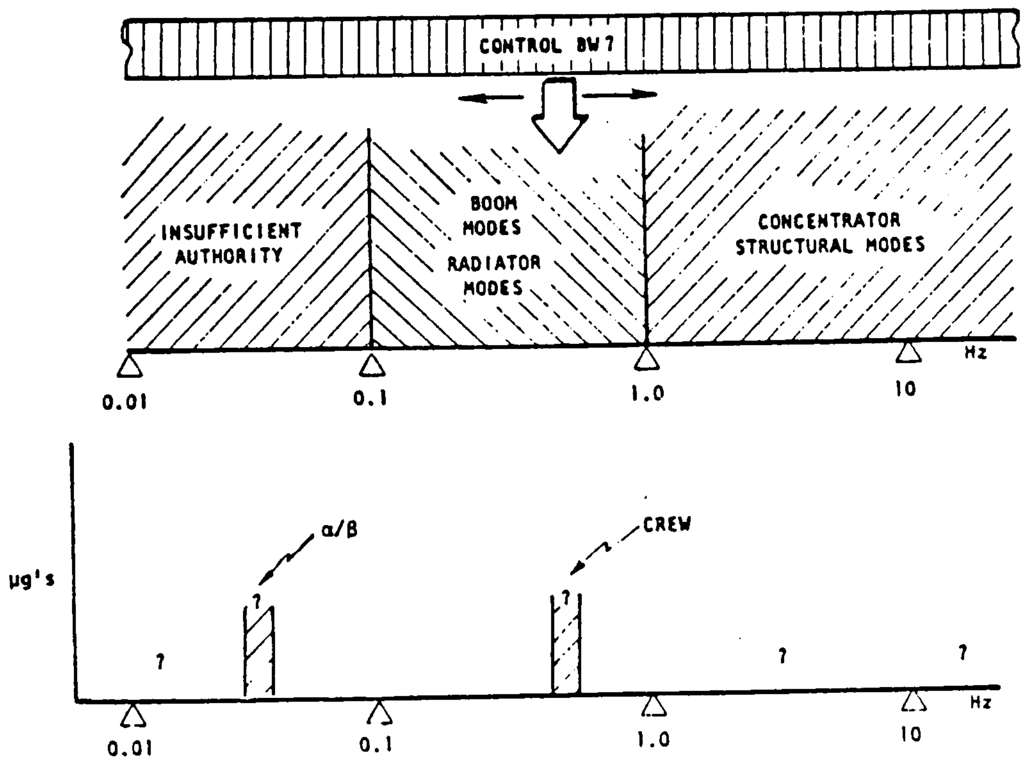

Figure 7.4-6 Control Bandwidth Issues

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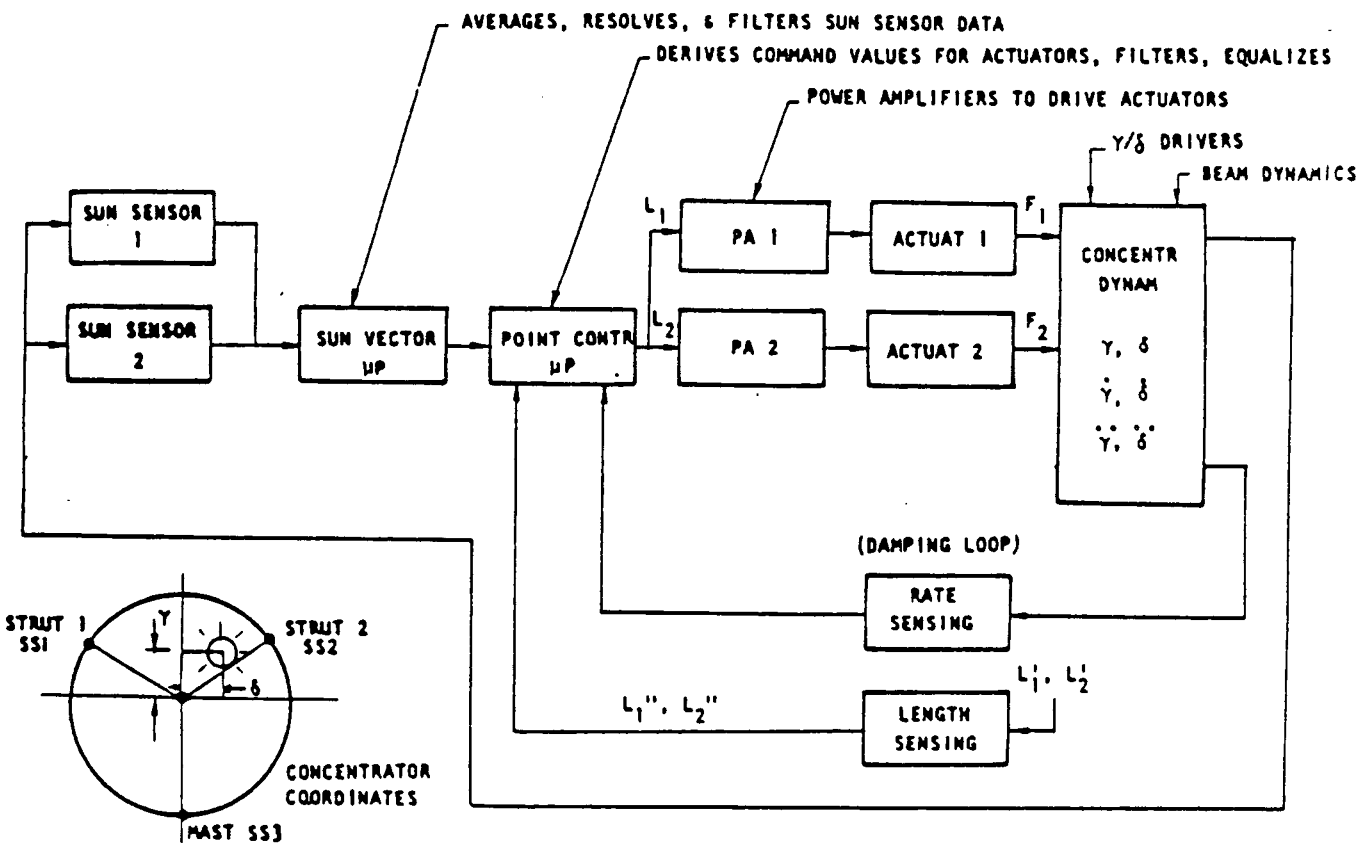


Figure 7.4-7 Fine Pointing Control Concept

concentrator about the pivot point of the reflector located at the reflector focal point or vertex, depending on the pointing concept. This rotation is accomplished by extending and/or shortening two linear actuators. As shown in Figures 7.4-3 and 7.4-7, equal motion of the struts causes rotation about the reflector's X axis; differential motion causes rotation about the Y axis. The resulting actuator displacements are fed back from position and rate sensors which are integral to the actuators. The two-axis fine pointing mechanism employing the dual ring-gimbal kinematics are slightly different in that the gimbal plane is not normal to the optical axis. Furthermore, the fine pointing control concept is simpler since length sensing is not required.

7.4.3.4 Fine Pointing Error Budget

An overall appreciation of the fine pointing accuracy requirement (0.1^{o}, at 3 sigma) may be gained by an examination of the gross pointing error in the alpha/beta joint control system. This gross error is estimated as 2^{o} per axis (3 sigma) resulting from the sources listed in Table 7.4-1.

Table 7.4-1
Estimated Pointing Error for Alpha/Beta Joints

Source	Error, deg
Knowledge of Sun Vector	0.1
Positioning of Alpha/Beta Joints	1.1
Mechanical Slop	1.0
Control Lag	0.5
Boom Distortion	1.7
RSS error at Receiver (2-axis)	2.0

Section 2.1.3 references a study which indicates that the alpha and beta joint contributions to this estimated error are conservative. From the above and the fine pointing requirement of 0.1^{o}, it is noted that the dynamic range for fine pointing is a factor of 20; normally considered a quite modest requirement. The current concern, however, is the uncertainty regarding the bandwidth of the gross pointing of the alpha/beta joints and the disturbances transmitted through the joints; i.e., what fraction of the 2.0^{o} is dynamic and how much of the dynamic fraction may reside above 0.1 Hz. At this time,

our judgment is that no significant disturbances will propagate to the beta support structure in the band above 0.1 Hz.

An initial allocation has been made for the pointing error of the FPCL; this allocation is given in Table 7.4-2.

Table 7.4-2
Fine Pointing Control System Error Budget
(3 sigma values)

Source	Basis	Error, deg
Sun Sensors	State-of-Art	0.02
Sun Sensor Alignment	Judgment	0.03
Concentrator Distortion	Judgment	0.04
Sun Vector Processor	Judgment	0.01
Initial Strut Alignment	Judgment	0.02
Strut Thermal Expansion	Estimate	0.01
Mechanical Creep	Judgment	0.01
Actuator Position Sensor	State-of-Art	0.01
Controller Processor	Judgment	0.03
Control Dynamics	Judgment	0.02
Total (1-axis)	Root-Sum-Square*	0.07
Total (2-axes)	Root-Sum-Square*	0.10

*Square root of the sums of the squares of each value

The sun sensor state-of-the-art has demonstrated a 3-sigma precision of less than 0.01 degrees which justifies the sensor budget established in Table 7.4-2. Alignment of the sun sensors (0.03°) refers to the uncertainty in knowing where the axes of the sun sensors are with respect to one another within the structural coordinate system. Unpredicted distortions in the concentrator structure could cause an apparent shift in the sensed sun vector (even if the overall effect on irradiance at the receiver were zero) and thereby create an estimated pointing error of 0.04°. Since this is estimated to be the single largest independent contributor to the pointing error, care must be taken to ensure the thermal-dimensional stability of the concentrator structure.

A conservative allowance of 0.01° has been made for approximations in the Sun Vector processor.

As previously stated, the previous reference control concept depended on knowledge of the lengths of the three columns (mast and two struts) supporting the concentrator. During initial alignment, it should be possible to adjust/calibrate the strut lengths to the equivalent of 0.02^{o} precision; this represents a magnitude of 3mm length uncertainty for the struts. The current reference control concept does not.

Following initial alignment and calibration, errors can accrue from thermal expansion and contraction and potentially from long-term (i.e., months) creep of the structure. Assuming low CTE (coefficient of thermal expansion) structural materials and design for the structure, it should be possible to limit the error source to less than the allocated 0.01^{o}. The creep allowance of 0.01^{o} is considered conservative.

The linear actuators include position transducers which will resolve length changes of less than 1 mm, a value which is within the allocation (0.01^{o}) made for this source.

Approximations and processing limitations in the controller processor are conservatively assumed to contribute 0.03^{o}.

Finally, an allowance of 0.02^{o} has been made for the combination of control system lags, overshoots, acquisition transients, shadowing biases, and unknown sources.

The above sources are considered to be single-axis values: mutually exclusive and stochastic. When these sources are combined, therefore, they root-sum-square to a single-axis budget of 0.07^{o}, and the two-axis value is the square root of 2 (1.4) times the single-axis value.

The allocations given above are believed to be achievable within practical design practices. However, detailed analyses are required during Phase C/D to verify this budget.

7.4.4 Optics Analysis

7.4.4.1 Introduction

Previous issues of DR02 documented preliminary modeling efforts used to evaluate the previous reference and two alternate pointing methods. On the basis of these limited optical performance analyses, it was concluded, at least tentatively, that any of the three configurations evaluated could be made to work. In ranking the optical performance of each configuration, the previous reference concept was selected.

7.4.4.2 Recent Activities

It was subsequently found, through more detailed optical modeling performed by GTRI, that the worst case spillage of the previous reference concept could be as high as 19% of the receiver power during the worst case corrected alpha pointing error state. This driver, along with the complexity of the proposed control loop and the structural softness, documented in section 7.3, drove the pointing concept toward the current configuration, described in section 2.2.3.

The current configuration is optically constrained in that the target receiver may rotate but not translate with respect to the CLC. Thus it is sufficient to evaluate the sensitivity of the receiver to ± 2 degree rotations about the focal point to characterize the non-zero pointing error optical performance of the concentrator.

A preliminary evaluation of the sensitivity of the receiver flux profile and intercept factor to alpha axis receiver rotation was performed by GTRI. Receiver rotations of 52 and 51.25 degrees with respect to the reflector optical axis were modeled. The results are shown in Figure 7.4-8. A 0.75 degree change in the receiver rotation decreased the receiver average power by a negligible 0.3%. In tilting the receiver from 52 to 51.25 degrees the minimum and peak flux are shifted but not increased or decreased.

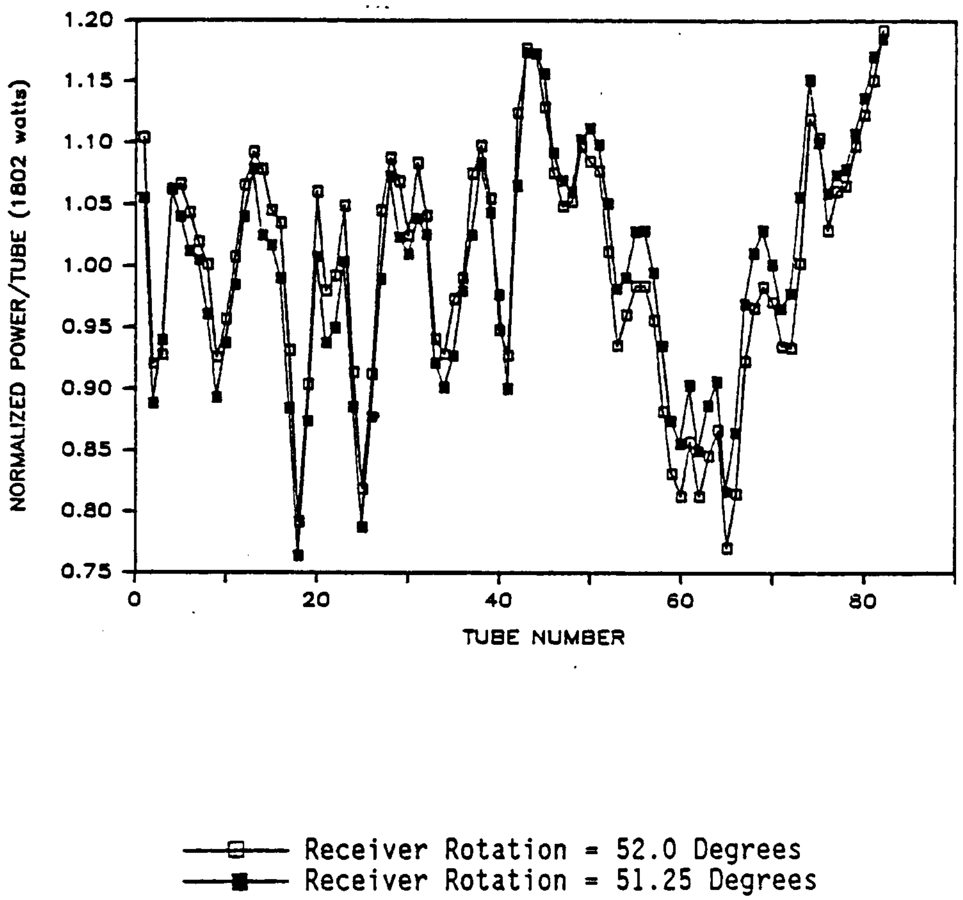


Figure 7.4-8 Reference Configuration Off-Nominal Optics Performance

7.4.4.3 Conclusions

The results of the current optical analysis indicate that the optical performance of the current reference is significantly better than that of any other pointing concept alternatives considered except the integrated concept. On the basis of the improved optical characteristics and upon significant advantages in natural frequency value, the 2-axis gimbal concept has been selected as the reference case.

7.5 CONCENTRATOR DEPLOYMENT TRADE STUDY

7.5.1 Introduction

Preliminary trade study was completed and documented in the June issue of DR02 which identified a feasible concept for the deployment of the hex-truss elements of the reflective surface of the concentrator. Based on the available requirements and constraints at that time, a power-pack-assisted, EVA deployment method was selected as the most cost effective approach.

However, since the conclusion of that study, new concerns have been raised, as part of the CETF activity, which indicated this trade should be revisited. more detailed trade study was completed which resulted in the selection of the current all latch erectable reflector configuration.

7.5.2 Alternatives Considered

Five alternate concepts were considered for the on-orbit assembly of the reflector subassembly. They included: a fully automatic, motorized, hinged/latched concept requiring no EVA for assembly; a fully deployable, non-motorized, hinge/latch concept requiring no EVA; a hinge/latch concept which is part EVA, part IVA assembly wherein all the panels are connected with hinges; a hinge/latch concept which is part EVA, part IVA assembly where the assembly of three groups of hex-trusses is required; and a latch only, all-EVA assembly concept. These concepts are designated A through E, respectively. They are illustrated in Figure 7.5-1. Concept D was the reference concept prior to this trade study.

7.5.3 Selection Criteria

The quantitative selection criteria used in the trade study are shown in Table 7.5-1. They include EVA and IVA resources required to assemble, overall program risk, and overall reflector subassembly program cost. Qualitative selection criteria, also shown in Table 7.5-1, were also used as

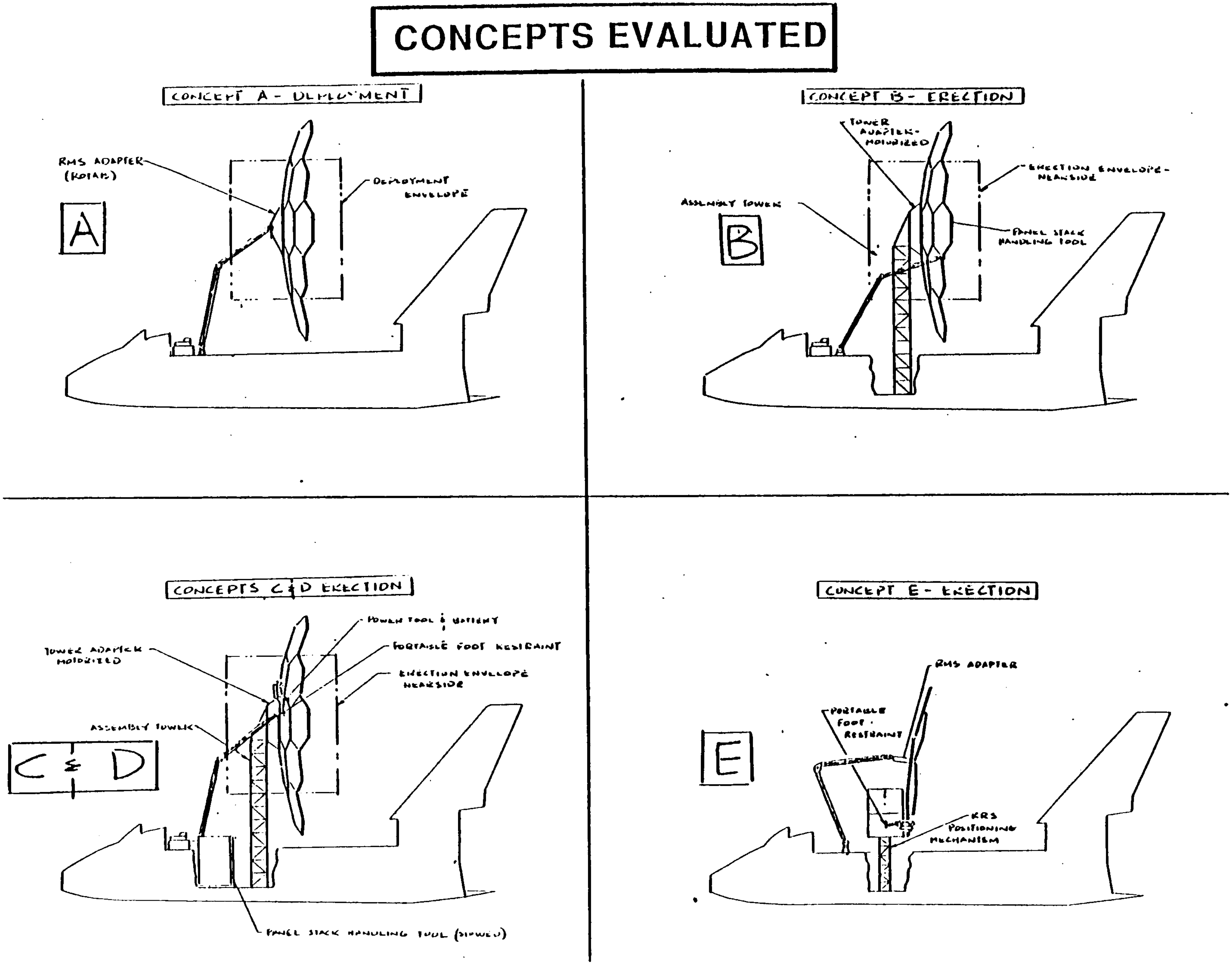


Figure 7.5-1 Alternate Reflector On-Orbit Assembly Concepts

Table 7.5-1
DEPLOYABLE/ERECTABLE REFLECTOR TRADE STUDY CRITERIA

Criteria	Weighting Factor
Quantitative Criteria	
EVA/IVA Resources Required	4
Overall Program Risk	3
Overall Program Cost	5
Qualitative Discriminators	
Ground and Flight Support Equipment Requirements	
Stowed Volume and Weight	
Deployment Tooling for Assembly, Integration and Checkout	
Restow Capability	
Neutral Buoyancy Simulator Compatibility	
Deployment Envelope	
Structural Stiffness	

discriminators. They include: ground and flight support equipment required, stowed volume and weight, deployment tooling for AICO, restow capability, NBS compatibility, deployment envelope, and structural stiffness. The weighting factors were established based on a negotiated understanding of the relative importance of the quantitatively evaluated criteria.

7.5.4 Evaluation Methodology

Each of the alternate assembly concepts was evaluated against the quantitative criteria by selecting one or more key parameters and using those as indicators. In the case of required EVA/IVA resources, a detailed timeline of EVA and IVA resource usage was developed for each concept. A summary of this analysis is shown in Table 7.5-2. In evaluating the overall program risk of each concept, the key parameters were complexity of the AICO, and ground test facilities, probability of successful restow after 30 year life, Neutral Buoyancy Simulator compatibility, and EVA time allowance criticality. A summary of the risk analysis is shown in Table 7.5-3. The evaluation of the overall cost was based on the estimated number of drawings required for the flight, flight support and ground support equipment included in each concept. This parameter has a good historical correlation to the relative program cost of

Table 7.5-2

EVA/IVA TIMELINE COMPARISON

CONCEPT	A	B	C	D	E
DEPLOY/ERECT ASSY TOWER	-	10.0(IVA)	10.0(IVA)	10.0(IVA)	-
REPOSITION RRS	5.0(IVA)	5.0(IVA)	5.0(IVA)	5.0(IVA)	5.0(IVA)
PREDEPLOYMENT OPNS	7.0(IVA)	14.0(IVA)	19 (EVA)	55 (EVA)	11.0(EVA)
DEPLOYMENT OPNS				14 (IVA)	
-PANELS	36.0(IVA)	131.9(IVA)	36.0(EVA)	36.0(EVA)	36.0(EVA)
-ROTATIONS @ ASSY(MAN)	-		9.5(EVA)	9.5(EVA)	6.0(EVA)
-UNLATCH PANELS	-	13.5(IVA)	4.5(EVA)	4.5(EVA)	-
-RMS TRANSLATIONS	-	5.8(IVA)	3.0(EVA)	5.5(EVA)	1.5(EVA)
TOTAL (EVA/IVA)	(0/48.0)	(0/180.2)	(72/15.0)	(110.5/29)	(54.5/5.0)
TOTAL (MAN)	48.0	180.2	87	139.5	59.5

Table 7.5-3

RISK BACKUP MATRIX

<table>
<tr><th>CONCEPT / PARAMETER</th><th>A</th><th>B</th><th>C</th><th>D</th><th>E</th></tr>
<tr><td></td><td colspan="4">o COMPLEXITY OF COUNTER BALANCE/OFFLOAD TOOLING IS SIGNIFICANTLY GREATER FOR ALL HINGE/LATCH CONCEPTS: DEPLOY FACEUP, FACEDOWN, AND IN THERMAL VACUUM CHAMBER</td><td rowspan="3">o INDIVIDUAL PANEL OFFLOAD/ C/B WITH NO MOTION</td></tr>
<tr><td></td><td colspan="2">o AUTOMATIC LATCH RELEASE, RESTOW, AND LOCKUP REQUIRED FOR NON EVA CONCEPTS - 30 YRS LIFE</td><td></td><td></td></tr>
<tr><td></td><td>o DRIVE UNITS MUST HAVE 30 YR LIFE FOR RESTOW</td><td></td><td></td><td></td></tr>
<tr><td></td><td colspan="5">o NBT REQUIRED BUT TANK DIMENSIONS LIMIT FULL SIZE TEST OF HINGED CONCEPTS</td></tr>
<tr><td></td><td>o NEUTRAL BOUYANCY TEST (NBT) LIMITED TO STRUT ATTACHMENT AND IVA</td><td>o NBT REQUIRED TO DEMON-STRATE RMS I/F'S AND CAPABILITY. COULD IMPACT DESIGN LATE IN PROGRAM</td><td></td><td></td><td>o REQUIRES LESS ENVELOPE FOR NBT</td></tr>
<tr><td></td><td></td><td>o RMS CAPABILITY DOUBTFUL</td><td>o EVA TIME CRITICAL</td><td>o EVA TIME CRITICAL</td><td>o EVA TIME CRITICAL</td></tr>
<tr><td></td><td colspan="2">o EDGE WEDGES MAY STILL REQUIRE EVA TIME</td><td></td><td></td><td></td></tr>
<tr><td>RANK</td><td>5</td><td>5 +</td><td>3</td><td>2</td><td>1</td></tr>
</table>

antennas produced by Harris Corp., the team member responsible for the reflector subassembly. A summary of the overall cost evaluation is shown in Table 7.5-4.

The evaluation of the qualitative criteria was used as a check to be sure that the quantitative criteria did not mask some important feature of any of the concepts. Concepts C and D are marginal with respect to existing neutral buoyancy simulation facility compatibility. Concept A is marginal with respect to restow capability. Concepts C and D are marginal with respect to EVA resource availability and allocation.

7.5.5 Results

The results of the evaluation are summarized in Table 7.5-5. Each of the concepts is evaluated against each of the criteria and ranked in the matrix. In the upper right hand corner of each EVA/IVA and Program Cost matrix cell the raw evaluation results are listed. The relative ranking of each concept for each criteria is located in the middle of each matrix cell. Rankings are from 1 to 5, 1 being the highest ranking. The criteria score for each concept is located in the lower left corner of each matrix cell. The criteria score is the product of the relative ranking and weighting factor, shown on the extreme right side of the matrix. The total score is shown across the bottom of the matrix. The lowest score indicates the highest ranked concept.

The concepts were ranked E, C, D, A, and B. Concept E was clearly superior to the other concepts. Concepts A, C and D are fairly close together, and concept B is significantly lower in the ranking.

7.5.6 Conclusions and Recommendations

The all latch concept appears to be clearly superior to the other alternatives considered. In addition to its high quantitative ranking, it is the most flexible with respect to assembly location and method, and has a reasonable assembly timeline. The all latch design is the recommended approach and has been included in the reference preliminary design concept.

Table 7.5-4

PROGRAM COST MATRIX

CONCEPT / PARAMETER	A	B	C	D	E
REFLECTOR	892	807	717	717	622
GROUND SUPPORT EQUIPMENT	525	460	330	330	255
FLIGHT SUPPORT EQUIPMENT	195	195	320	320	185
TOTAL	1612	1462	1367	1367	1062
RELATIVE VALUES	1.51X	1.38X	1.29X	1.29X	1X

NUMBERS ARE COMPLEXITY FACTORS BASED ON THE DRAWING COUNT FOR EACH SYSTEM.

Table 7.5-5

REFLECTOR ASSEMBLY EVALUATION MATRIX

CONCEPT / PARAMETER	A	B	C	D	E	WEIGHTING FACTOR
EVA/IVA	0/48 1 4	0/180.2 4 16	72/15 3 12	110.5/29 4 16	54.5/5 2 8	4
OVERALL PROGRAM RISK	5 15	5+ 15	3 9	3 9	1 3	3
PROGRAM COST	1.5X 5 25	1.4X 4 20	1.3X 3 15	1.3X 3 15	1X 2 10	5
TOTALS	44	51+	36	40	21	

If, in the future, additional concern as to the availability of EVA resources precludes the selection of the all-latch design, then concept A, an all automatic deployment, could be utilized. It is also recommended that other automatic deployment schemes, such as robotic assembly be considered before this contingency is exercised.

7.6 ORC Radiator Method of Heat Rejection Trade Study

Results of an earlier radiator trade study (Number 81) were reported in Section 7.1.8 of DR19 DP4.3. Three different ORC radiator concepts were evaluated at that time: 1) a constructible radiator with round interfaces, 2) a constructible radiator with a single, flat interface and 3) a deployable radiator using low capacity heat pipes.

The second concept was selected for the preliminary ORC radiator design and a description of the current configuration appears in Section 2.2.4 of this document. The basis for this selection considered the following factors:

1. Comparative cost
2. Commonality with other Space Station systems
3. Lowest weight
4. Best match to flat ORC condenser interface
5. Concept adapts to a constant temperature (condensing) ORC waste heat rejection cycle
6. Minimal departure from original NASA reference design concept

The comparative cost analysis considered hardware manufacturing assembly, reboost (radiator area) and launch (radiator weight) costs.

This section presents the results of an additional trade study to evaluate alternative radiator designs for use with the ORC PGS. The study includes a deployable pumped loop radiator concept and two different versions of a constructible radiator design utilizing double-sided, flat interfaces. The pumped loop design concept will be discussed first.

Deployable Pumped Loop Radiator

The current trade study to evaluate a deployable, pumped loop radiator concept for the ORC was conducted to provide an alternative design to the constructible heat pipe radiators. The recommendation for the selection of an ORC radiator design, made upon the basis of these studies, is given in the final part of this section.

Figure 7.6-1 shows schematically a pumped loop radiator design consisting of ten pumped liquid honeycomb panels, plumbed with a primary and redundant flow subsystem. Each loop contains a single accumulator and redundant pumps. Toluene was selected as the pumped loop heat transfer fluid.

An optimization study was conducted to determine a minimum weight radiator design consistent within coolant flowrate and subsystem pressure drop requirements. The results of this study are shown in Figures 7.6-2 and -3 as a function of the assumed pumped liquid temperature drop for various tube diameters.

The effect of temperature drop (in the pumped fluid as it flows through the radiator) on the weight of the panels and the total subsystem weight is shown in Figure 7.6-2. The latter includes a subsystem weight penalty that must be added to compensate for the extra power consumed by the fluid pumps.

From this parametric analysis a single point design was chosen that corresponds to a radiator delta-T of 299K (80F) and a tube internal diameter of 0.25 cm (0.100 in). This particular selection represents a compromise in pump capacity and the associated pressure drop requirements. As shown in Figure 7.6-3, the flowrate increases as the allowable temperature delta-T decreases, producing a smaller radiator area but with significant increases in the total subsystem weight (Figure 7.6-2).

Details of the pumped loop panel design are given in Figure 7.6-4. Each of the ten panels is 8.0 by 2.3 m (26.4 by 7.5 ft) and 1.8 cm (0.67 in) thick. The panels are constructed by incorporating tube extrusions into a honeycomb structure which is sandwiched between two face sheets of 0.25 mm (0.01 in) aluminum. The tube extrusions consist of an 1.3 mm (0.050 in) bumper and 0.5 mm (0.020 in) wall, with 2.54 mm (0.100 in) ID tubes.

A summary of the pumped loop design characteristics is given in Table 7.6-1a. Toluene, used as the radiator pumped heat transfer fluid, flows at a rate of 1.57 kg/sec (12,398 lb/hr). Total planform area for the deployed panels is 175 m^2 (1880 ft^2). The panels weigh 988 kg (2155 lb), giving a specific panel weight of 5.6 kg/m^2 (1.15 lb/ft^2). Total weight of the pumped loop radiator, designed for the ORC application, is 1403 kg (3086 lb).

Alternate Constructible Radiator Concepts

The baseline ORC radiator design described in Section 2.2.4 is a constructible concept that utilizes multiple heat pipe panels, each of which interface with the ORC condenser on the same side. In order to decrease the overall length of the condenser (and therefore subsystem weight), two different design configurations were evaluated for concept feasibility. Both of the concepts utilize alternate sides of the ORC condenser for the contact interface, thus providing the opportunity to shorten its overall length.

Concept 1, shown in Figure 7.6-5, utilizes straight heat pipe radiator panels, aligned 45° with respect to the ORC condenser. All adjacent panels interface with one side of the condenser, while opposite panels interface with the other side.

A design layout problem was encountered with this concept. Due to the proximity of the overlayed panel interfaces, it was not possible to provide the necessary clearance to install the required number of structural bolts to ensure the needed contact pressure. Accordingly, Concept 1 was not pursued further.

An alternative layout is shown in Figure 7.6-6. For Concept 2, each panel contains a 45° bend, near the heat pipe adiabatic sections. During installation, the heat pipe evaporator sections intersect the ORC condenser at right angles. For this configuration, all adjacent panels interface with alternate side of the condenser.

Sufficient clearance is available to facilitate the removal of individual panels on-orbit and for the initial assembly sequence. The design provides a potential reduction in condenser weight of about 4.1 kg (93 lbs) and 4.1 m (13.2 ft) in overall length. Concept 2 represents a viable alternative to the

Table 7.6-1 a

ORC Radiator Preliminary Design Characteristics

MODULE SIZE: 25 kWe

TYPE: Deployable pumped liquid

HEAT REJECTION: 113.3 kWt

HEAT TRANSPORT LOOPS: (primary and redundant loops used)

FLUID - Toluene

FLOWRATE - 12,398 lb/hr

DELTA P - 23 psi

POWER USE - 898 watts (based on Moog 50-498 Pump)

PANEL SIZE: 2.3 m (7.5 ft) in) x 8.0 m (26.4 ft) in)

NUMBER OF PANELS: 10

PANEL PLANFORM DEPLOYED AREA: 175 m^2 (1880 ft) in^2)

MATERIAL: Extruded bumpered aluminum flow tubes with aluminum fins

WEIGHT:	Panels	980 kg (2155 lb)
	Deployment mechanisms	296 kg (651 lb)
	Pump packages (4 required)	56 kg (123 lb)
	Flex hoses (including fluid)	31 kg (68 lb)
	Disconnects (including fluid)	7 kg (16 lb)
	Plumbing (tubing, valves, etc.)	32 kg (70 lb)
	Plumbing enclosure	1 kg (3 lb)
	TOTAL	1403 kg (3086 lb)

COATING: Z93 Zinc Oxide Paint

T-ENV: 213 K (-76F) (sink temperature)

T-COOL INLET: 340 K (152F)

T-COOL OUTLET: 295 K (72F)

current baseline heat pipe radiator concept described in Section 2.2.4.

A side view of the double-sided, contact interface concept is shown in Figure 7.6-7. Insertion guides are provided along both edges of the condenser to assist in the installation of individual heat pipe panels. After the panels are in place, each interface is secured with a separate pressurized diaphragm to affect the proper thermal contact conductance.

ORC Pumped Loop vs Heat Pipe Radiator Trade Study

An optimization study, conducted to assist in the selection of the ORC radiator preliminary design, included a comparative evaluation of the deployable pumped loop vs the current baseline heat pipe radiator design. The two concepts were compared quantitatively on the basis of life cycle cost, IOC costs, mass and area considerations, and on a "relative cost" basis which considered only reboost and launch costs. They were compared qualitatively on the basis of development risk, operational considerations, STS integration, commonality, reliability and cycle match.

The study ground rules, as listed in Table 7.6-1b, were established to ensure that the two designs were compared on an equivalent basis. Operating state points, environmental conditions, and the basic cost rates were identical for the two systems.

TABLE 7.6-1b. ORC Radiator Study Ground Rules

1. Identical module size (25 kWe net)
2. Identical cycle state points
3. Identical sink temperatures (-76°F)
4. Identical coating properties (ϵ= 0.900)
5. Identical drag/reboost scenario (\$1145/m^2-yr)
6. Identical station buildup scenario (75 kWe - 300 kWe)
7. Identical EVA, IVA and MRMS rates (\$103K, \$15K, \$0)

The panel design presented in Figure 7.6-4 formed the basis for the pumped loop radiator. The heat pipe design was based on the characteristics given in Table 2.2.4-1 using the Lockheed tapered artery heat pipe shown in Figure 2.2.4-2. For consistency, the development and procurement costs were based upon LTV estimates, using their space radiator experience.

The results of this study are presented in Table 7.6-2.

TABLE 7.6-2. Radiator Study Comparisons

Parameter	Pumped Loop Radiator	Heat Pipe Radiator
Life Cycle Cost	\$613 Million	\$753 Million
Life Cycle Cost*	\$526 Million	\$562 Million
IOC Cost	\$ 78 Million	\$ 87 Million
Weight, kg (lb)	1403 (3086)	1981 (4359)
Area, m^2 (ft^2)	175.0 (1880)	159.8 (1719)
Pump Power, kW	898	0

*Assumes no radiator replacements at station EOL (30 years).

On the basis of the trade study, the pumped loop radiator provides the following advantages relative to the heat pipe concept.

1. Lower LCC
2. Lower IOC cost
3. Lower technical risks
4. Minimum start-up problems
5. Better packaging
6. Commonality with fluid management components
7. Lower weight

Similarly, the heat pipe radiator concept was found to have certain advantages over the pumped loop design:

1. Better overall reliability
2. No parasitic power requirements
3. Graceful degradation
4. Maximum commonality
5. Potential utilization of current ADP technology
6. Lower area
7. State point match

Although both radiator concepts provide some level of commonality with the other Space Station work packages, maximum commonality would be achieved with the constructible heat pipe radiators.

In addition, a simplified analysis technique was developed to compare the relative significance between radiator weight and area as a selection criteria. The parameter chosen for this analysis has been identified as "relative cost" and is defined:

$$\text{Relative Cost} = (\text{Area})*(0.2)*(\$567/\text{ft}^2) + (\text{Weight})*(\$3050/\text{lb}).$$

Radiator area produces a cost penalty for the Space Station and is reflected in propellant reboost costs, associated with increased drag in the direction of the velocity vector. Similarly, radiator weight produces an additional penalty via increased launch costs. The sum of these two separate cost factors (reboost and launch) have been identified as "relative cost", and the calculated values plotted in Figure 7.6-8 for several specific radiator designs.

Relative cost represents only some small fraction of the total LCC since such factors as hardware development, procurement and on-orbit maintenance have not been included. The other parameter plotted in Figure 7.6-8 is specific weight, and is defined as the ratio of total radiator weight to its area.

The radiator design points plotted on Figure 7.6-8 are identified below:

1. Pumped loop radiator without power penalty
2. Pumped loop radiator including power penalty
3. Heat pipe radiator with flat interface
4. Heat pipe radiator with quick disconnect interfaces
5. Heat pipe radiator using ADP technology

As shown, the magnitude of the relative cost associated with the pumped loop concept depends upon whether or not the penalty associated with the parasitic power requirement of the pump are considered. Although the pumped loop radiator panels can be fabricated with less weight, the added weight penalty must nevertheless be launched to orbit. With this penalty taken into account, the relative cost of the ORC pumped loop concept compares closely with that calculated for the heat pipe radiator with the single-sided, flat interface.

The potential additionally exists to reduce the relative cost of the heat pipe radiator by replacing the pressurized, flat interface with the multiple, quick disconnect-type of interface. As shown on Figure 7.6-8, this could result in a lighter weight radiator and a lower relative cost.

The last data point represents an ORC radiator design incorporating the aluminum, dual-slot heat pipe currently being evaluated under the NASA ADP. Although this design has potential advantages, it must be considered as a higher risk option at this time.

The data presented in Figure 7.6-8 are summarized below in Table 7.6-3.

TABLE 7.6-3. Relative Cost of Potential ORC Radiator Designs

Concept	Specific Weight (lb/ft^2)	Area ft^2	Relative Cost $M*
Pumped Loop-no power penalty	1.64	1880	9.6
Pumped Loop-w/power penalty	2.16	1880	12.6
Heat Pipe-Flat Interface	2.53	1719	13.5
Heat Pipe-QD Interface	2.32	1567	11.2
Heat Pipe-ADP Technology	1.29	1567	6.3

*Considers reboost and launch costs only.

Based upon the results generated in this trade study, the heat pipe radiator concept remains as the current choice for the ORC radiator.

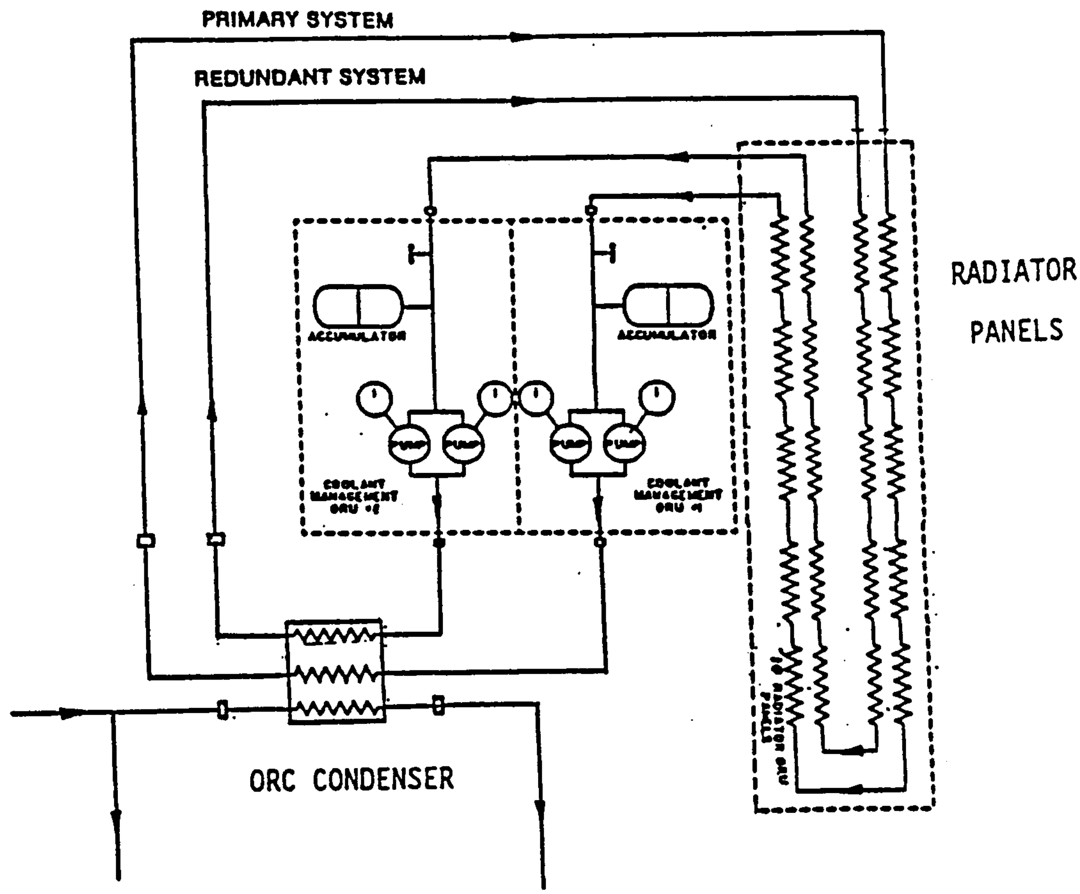


Figure 7.6-1. ORC Deployable Pumped Loop Radiator Schematic

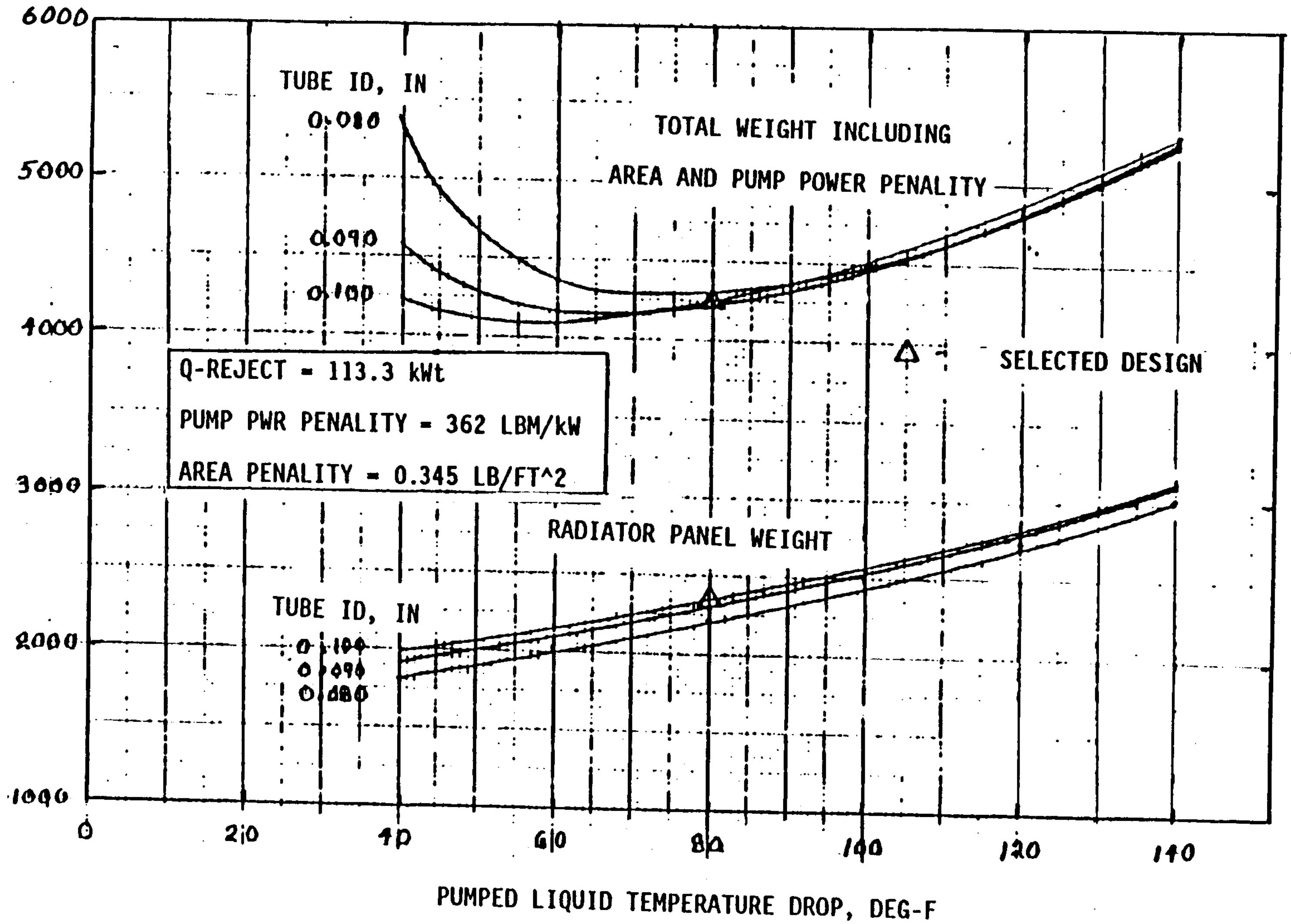


Figure 7.6-2. Total Weight and Panel Weight versus Pumped Loop Temperature Drop

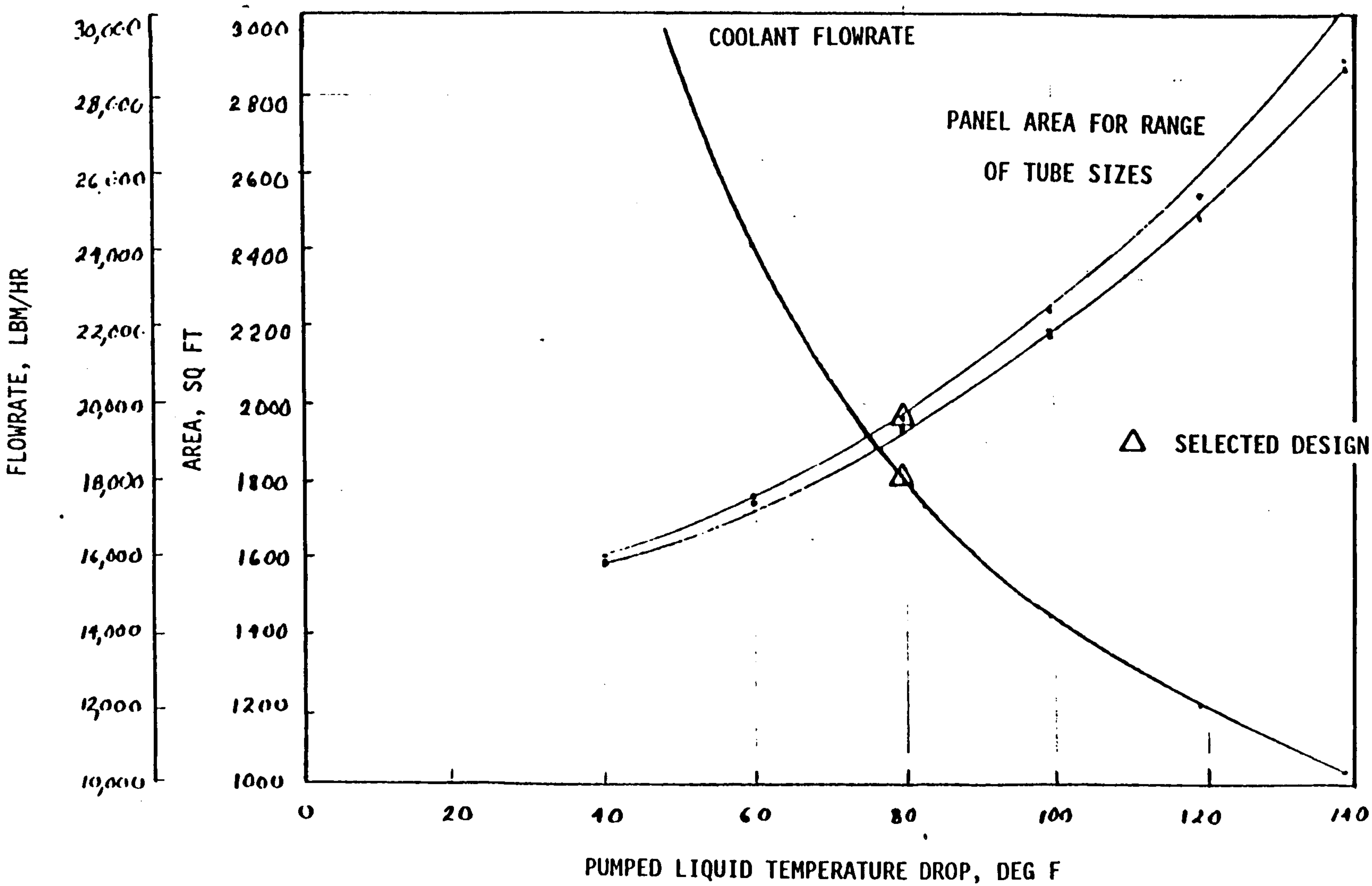


Figure 7.6-3. Flow Rate and Area versus Pumped Liquid Temperature Drop

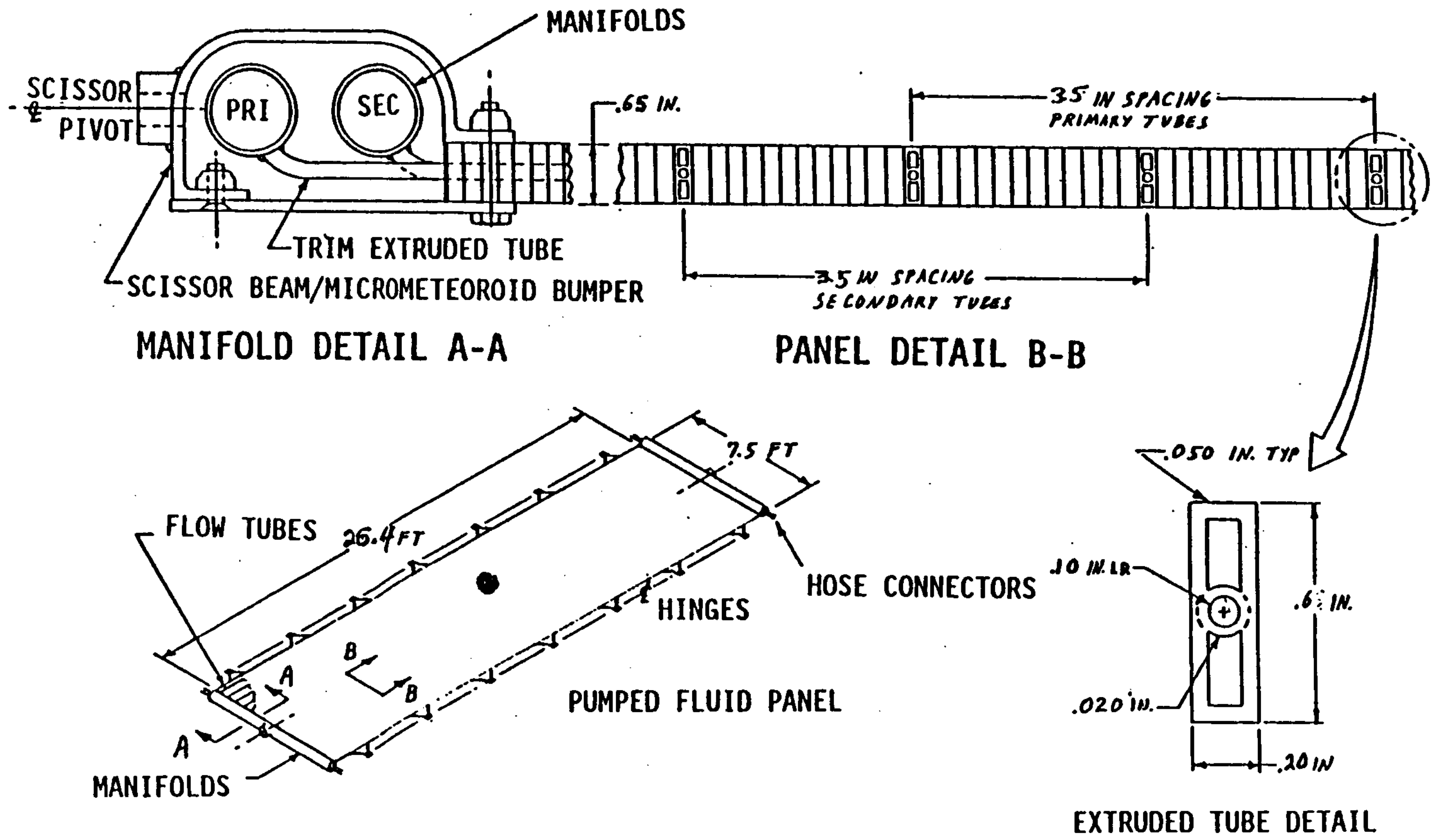


Figure 7.6-4. ORC Pumped Loop Radiator Panel Design

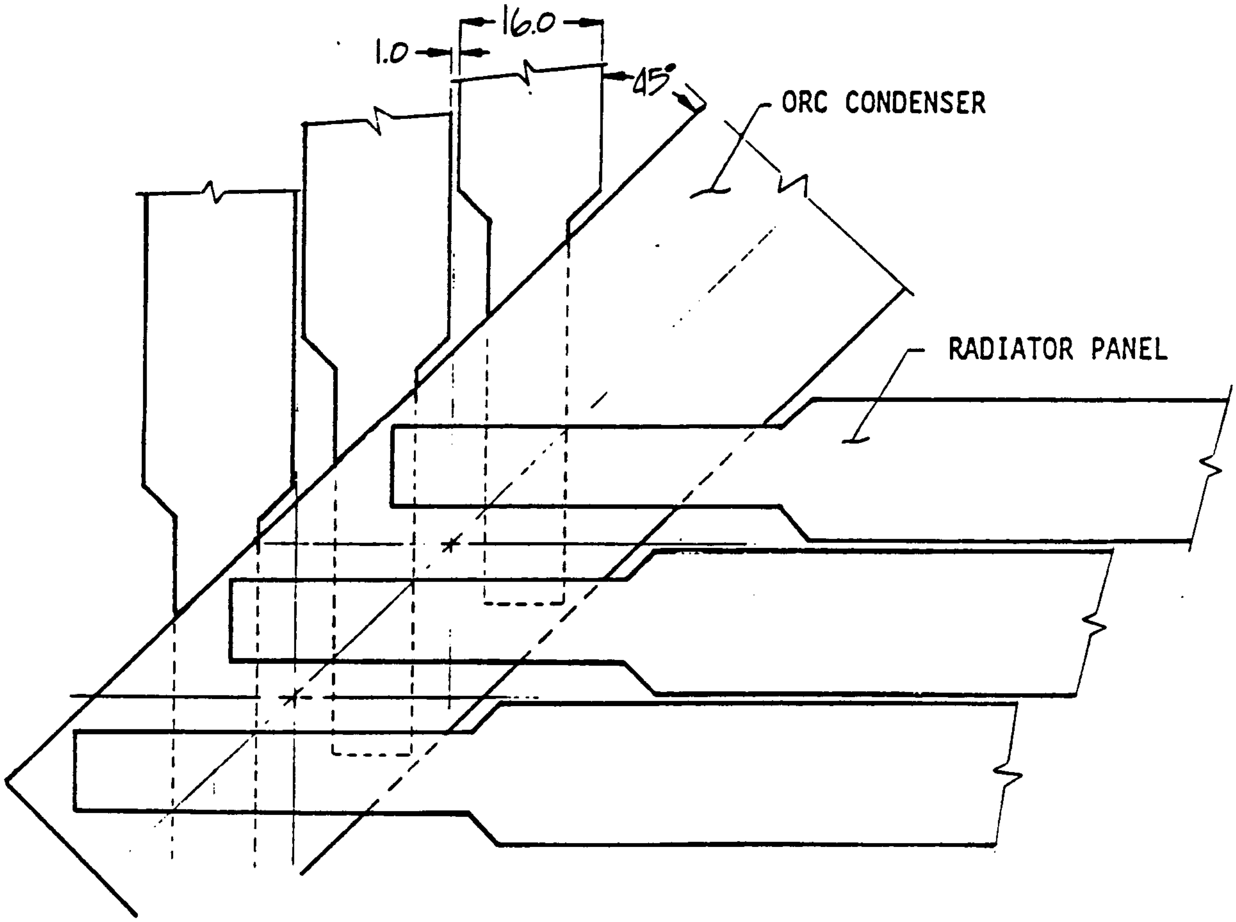


Figure 7.6-5. Concept 1, Diagonal Condenser Concept with Straight ORC Radiator Panels

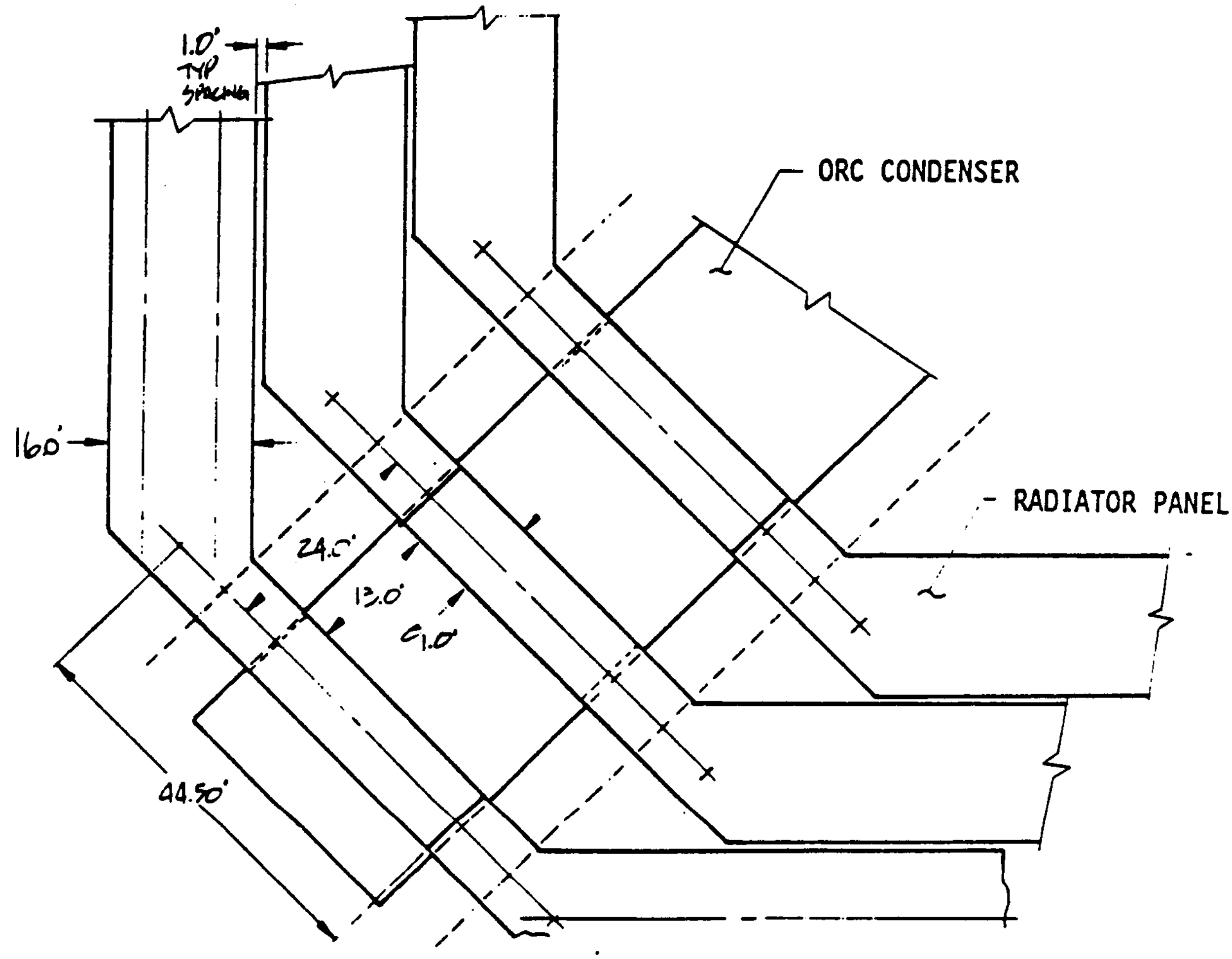


Figure 7.6-6. Concept 2, Diagonal Condenser Concept with Bent ORC Radiator Panels

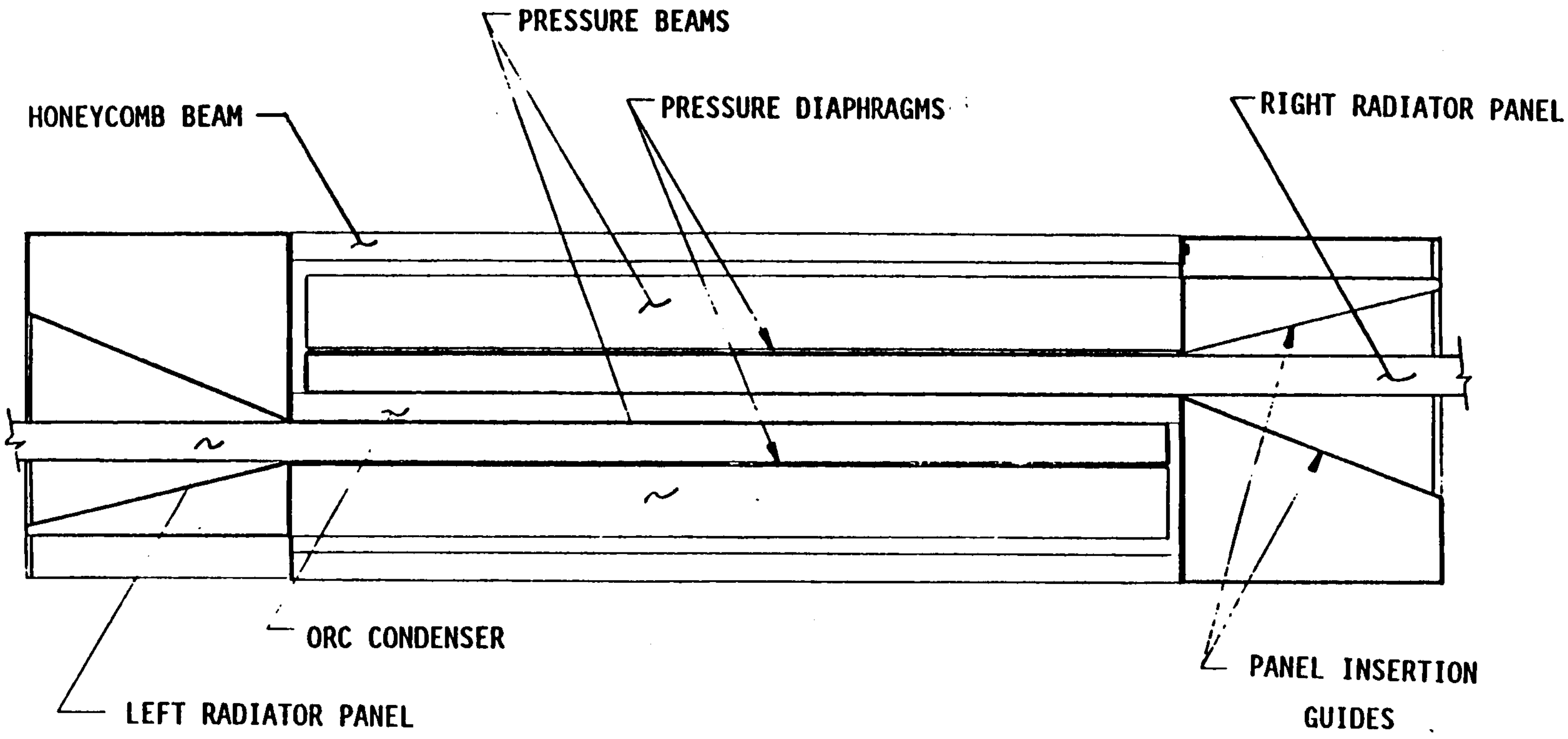


Figure 7.6-7. Details of Diagonal Condenser Concept, Two-Sided Interface

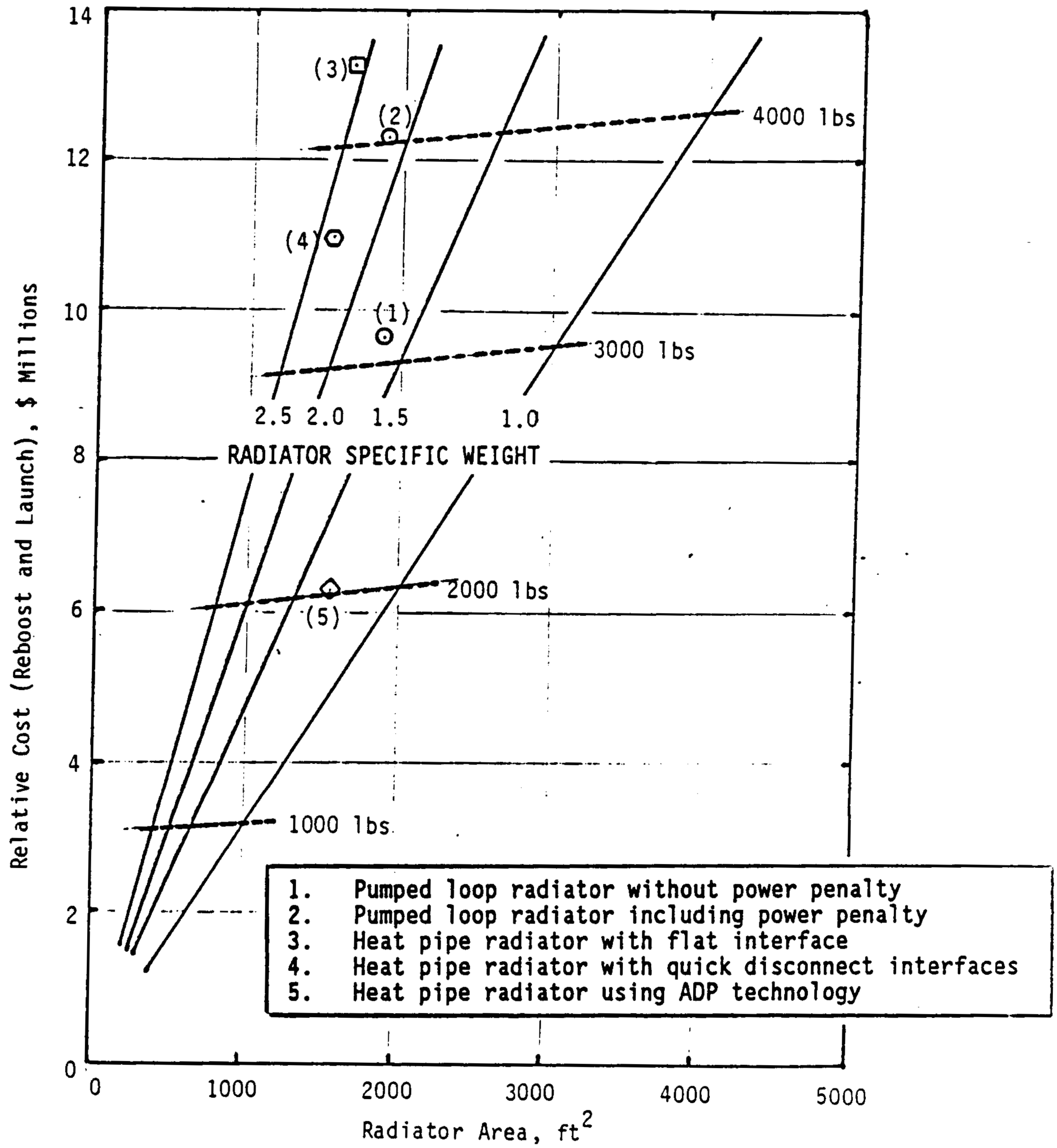


Figure 7.6-8. Relative Cost Comparison for Different ORC Radiator Concepts

7.7 Radiator Location

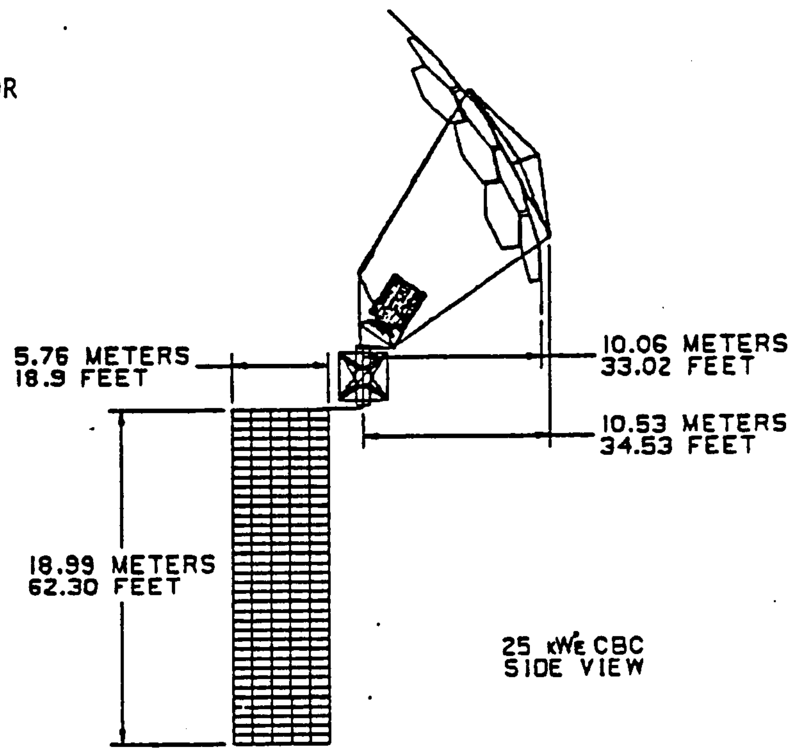
Two options were considered for the radiator location: collocated and underslung, as shown in Figure 7.7-1. The reference configuration has been changed to a collocated configuration as a result of this study. This configuration results in a less complex structure, lower mass, better deployability, and elimination of scarring for growth at the penalty of moving the center of gravity (for IOC only) off the alpha axis. The resulting micro-g effect and bending loads are considered acceptable. Table 7.7-1 summarizes the principal characteristics of the two options.

The reduced plumbing required for the collocated radiator results in many advantages. Because the plumbing length is considerably shorter, fluid pressure losses and mass are reduced. The plumbing in the underslung position requires beta joint accommodation with the potential need for quick disconnects on both ends of the bay. Collocated, it is possible to launch and deploy the receiver, PCU and radiator in a single launch package and automatically deploy. This was feasible for CBC but not for the ORC radiator boom which is still a separate launch package.

The underslung position requires scarring for growth. When the second SD power module is added at the beta joint location, the underslung radiator must be disconnected and reconnected in the collocated position This would require shutting down the power module during growth and require additional EVA time. The reduced sink temperature in the underslung position could not be utilized in design because of the eventual conversion of the module to the collocated position in growth.

The major disadvantage of the collocated configuration is the change in the center of mass (c.m.). The c.m. remains on the beta axis of rotation but moves outward 26 feet and 24 feet from the alpha axis of rotation for CBC and ORC, respectively. The 360^{0} rotation of the alpha joint during every orbit

UNDERSLUNG RADIATOR



COLLOCATED RADIATOR

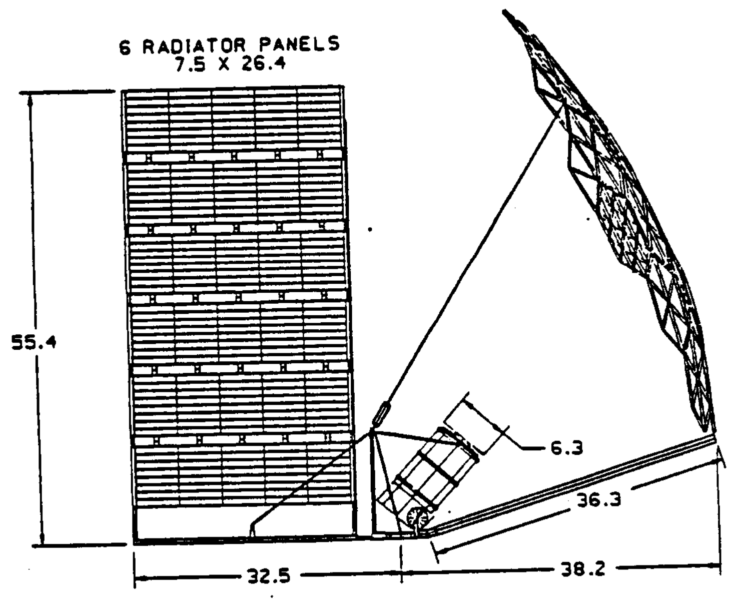


Figure 7.7-1

Radiator Location Options

TABLE 7.7-1

COLLOCATED RADIATOR VERSUS UNDERSLUNG RADIATOR

Feature	Collocated	Underslung
Plumbing	x 3m (10 ft) to radiator x Less complex	o 7.6m (25 ft) to radiator o Requires Beta joint accommodation
Parasitic Losses	x Reduced fluid pressure drop	
Mass	x Less due to less plumbing	
Deployability	x Deployable interface structure & plumbing (CBC only)	o Erectable interface structure and plumbing
Growth	x No scarring	o Growth requires moving radiator to collocated position when second module is added at Beta joint
Sink Temperature	o Higher sink temperature	x Lower sink temperature but only at IOC
Center of Mass	o CBC: 12,000 lbs located 26 ft from axis of rotation o ORC: 13,700 lbs located 24 ft from axis of rotation	x Located at and axis of rotation
Truss Bending Loads	o Increased load located at module: CBC: 0.013 lb ORC: 0.014 lb	x Base case
Micro G Effect	o Increases station microgravity by CBC: 0.59×10^{-7}g ORC: 0.62×10^{-7}g (Station allowable is 10^{-5}g)	x Base case

x Means Preferred Option

results in a centrifugal force. For CBC, a 12,000 lb module results in a 0.013 lb load located at each end of the transverse boom that would create bending loads in the truss. Correspondingly for ORC, a 13,700 lb module results in a 0.014 lb load. The module rotation results in a change in the station center of mass. For a 450,000 lb station, there is a 0.59×10^{-7}g and 0.62×10^{-7}g microgravity effect for CBC and ORC, respectively. This is less than 1% of the station allowable. This microgravity effect will only occur in the single module configuration. When the second module is added during growth these loads are eliminated.

7.8 CBC PUMPED-LOOP VERSUS HEAT-PIPE RADIATOR TRADE STUDY

Pumped-loop and heat-pipe radiators for a CBC power system have been compared according to selection criteria established by NASA-LeRC (1). By these criteria, a pumped-loop radiator is preferable to a heat-pipe radiator. The IOC cost for heat-pipe radiators would be greater than the IOC cost for pumped-loop radiators by 13%, the life-cycle cost would be greater by 40%, and the EVA time required for maintenance would be greater by a factor of about 10. The latter number is conservative based on more recent studies of time and motion for the general class of space erectable radiators.

7.8.1 Methodology

The characteristics of heat-pipe and pumped-loop radiators were compared in order to determine which design should be incorporated in the CBC power system. A previous study (2) had determined that, for this application, pumped-loop radiators were superior to pumped-loop/ heat-pipe hybrids. In the present study, the pumped-loop and direct heat-pipe designs were compared quantitatively on the bases of life-cycle cost, IOC cost, mass, area, and maintenance requirements; and qualitatively on the bases of complexity, development risk, cycle match, power module integration, and STS integration. These criteria address most of the concerns expressed by NASA (1).

The ground-rules, as listed in Table 7.8-1, were established to ensure that the two designs were compared on equivalent bases. Operating points, environmental conditions, and basic cost rates are identical for the two systems, with the exception that the pumped-loop design includes the pumping house-load. The assumed CBC State-Point Diagram is shown in Figure 7.8-1. Life-cycle costs are calculated assuming that the growth in demand for solar-dynamic power follows the scenario in Table 7.8-2.

The design shown in Figure 7.8-2 (3) formed the basis for the pumped-loop system. The heat-pipe radiator design was based on a Grumman dual-slot heat pipe shown in Figures 7.8-3 and 7.8-4 (4,5). For consistency, development and procurement costs for both designs are based on estimates by LTV (2,3,6,7), which has experience in producing space radiators. The development and procurement costs used in the study represent Rocketdyne's costs to NASA. All costs are in 1987 dollars.

For the pumped-loop radiator, life-cycle cost was minimized by optimizing the designs of the radiator and coolant management ORUs. The radiator panels were optimized by trading mass versus puncture reliability. The results of this optimization are shown in Figure 7.8-5. The abscissa of Figure 7.8-5 relates panel life-cycle cost to the 10-year, single loop puncture reliability of a radiator sized to reject 67.4 kW. Figure 7.8-6 relates the nominal puncture reliability to the actual lifetime (20 year) reliability of the pumped-loop radiator panels. The coolant management ORU was optimized by varying the number of redundant pumps. Life-cycle cost is minimized when a single pump is specified, but a second pump is necessary to meet fail-operational requirements. For the heat-pipe radiator, a panel failure rate of 1.62 E-6 failures per hour is assumed. This rate is a composite of 7.1 E-7 failures per hour due to meteoroid punctures and 9.1 E-7 failures per hour of the heat-pipe quick-disconnect (5,8).

Basic cost rates (for transportation, reboost, and EVA and IVA time) are consistent with the bases of the economics studies contained in previous DR-19 data packages (2). Component installation and replacement times were estimated by Rocketdyne, and are consistent with the values of NASA (9). Total costs for the IOC and the life-cycle are broken down to show the contributions of development, procurement, transportation, installation or maintenance, and reboost. The space station is assumed to have a 30 year design (depreciation) life, with component lifetimes extended indefinitely through ORU replacement, as has been assumed in previous economic studies (2).

7.8.2 Results

Parameters describing the pumped-loop and heat-pipe radiator designs are presented in Tables 7.8-3 and 7.8-4 respectively. The parameter values were representative of the two designs at the time the study was done. Both designs are for power systems providing 25 kWe net, plus 33% peaking. Quantitative comparisons between the two designs are presented in Table 7.8-5, while qualitative comparisons between the designs are presented in Table 7.8-6.

On the basis of life-cycle cost, the pumped-loop concept is preferred to the heat-pipe concept, the difference in costs being nearly $230 Million. On the basis of IOC cost, the pumped-loop concept is again preferred, the difference in costs being nearly $9 Million. These cost comparisons, with breakdowns, are shown in Figures 7.8-7 and 7.8-8. The IOC costs for the pumped-loop radiator include the costs of on-orbit spares for each of the three radiator ORUs. IOC costs for both designs are calculated for systems on orbit and installed.

On the basis of mass, the pumped-loop concept is preferred, as the mass of 11 heat-pipe radiators on the growth station would be about 3,930 kg (8,670 lbm, or 31%) greater than the mass of 12 pumped-loop radiators (including one spare). On the basis of drag area, however, the heat-pipe concept is preferred. The planform areas of the two designs are shown in Table 7.8-5.

On the basis of maintenance time, the pumped-loop concept has a significant advantage. As shown in Table 7.8-5, maintaining the heat-pipe radiators over the life cycle requires about a factor of 10 times more EVA time (740 additional EVA hours) than is required for maintaining the pumped-loop radiators. The difference in module downtime is less pronounced, however, since an additional 24 hours of downtime is assumed for each of 18.5 pumped-loop ORU failures. The MRMS time required to maintain the pumped-loop radiators is negligible in comparison with the MRMS time required to maintain the heat-pipe radiators. Figure 7.8-9 summarizes the differences in limited resource utilization required for the maintenance of the two designs.

On the basis of complexity, the heat-pipe concept is preferred since it has both fewer ORUs and a smaller number of unique sets of parts. The pumped-loop concept is preferred on the basis of development risk because it has an STS heritage, whereas the direct-loop/heat-pipe concept employs technologies not previously used in space, particularly for manned systems. On other qualitative issues, the pumped-loop concept is thermodynamically better suited to cycles (such as the CBC) with sensible heat rejection, while heat pipes are better suited to cycles (such as the ORC) with constant temperature heat rejection. The pumped loop concept minimizes the exposure of the CBC working fluid loop to risk of meteoroid puncture (by avoiding the extension of the working fluid boundary along the length of the radiator), more readily accommodates active cooling of the power system electronics (if necessary), and is more easily packaged in the STS cargo bay.

7.8-3 Conclusion

On the basis of a variety of criteria, a pumped-loop radiator appears to be better than a heat-pipe radiator for a CBC power system. Only on the bases of complexity and drag area does the heat-pipe radiator hold an advantage. By more significant criteria, such as life-cycle cost and life-cycle maintenance time, the advantage is decisively in favor of the pumped-loop concept, which is therefore Rocketdyne's choice at this time for the CBC power system radiator.

References:

1) Telefax, J. Calogeras and S. Simons (NASA-LeRC) to G. J. Hallinan (Rocketdyne), concerning Solar Dynamic questions/criteria, 29 April 1986.

2) "Time-Phased SE&I Study Products, DR-19, D.P. 4.4", Rocketdyne, Canoga Park, 19 November 1985.

3) Telefax, R. L. Cox (LTV) to Rocketdyne, "Letter Progress Report for Solar Dynamic Heat Transport and Radiator Trade Studies, P.O. R50PAB85821601", dated 17 December 1985, Sent 7 January 1986.

4) Personal Communication, C. T. Kudija and E. C. Hylin (Rocketdyne) to J. Cassidy, K. McLannin, and C. Ensworth (NASA-LeRC), regarding NASA-LeRC/Grumman design for CBC heat pipe radiator, 27 March 1986.

5) Personal Communication, C. T. Kudija and E. C. Hylin (Rocketdyne) to A. W. Carlson (Grumman), regarding reliability of Grumman heat pipes for CBC heat pipe radiator, 24 April 1986.

6) Personal Communication, C. T. Kudija (Rocketdyne) to M. Fleming (LTV), concerning costs of radiator hardware, 5 February 1986.

7) "Solar Dynamic Heat Transport and Radiator Design Trade Studies, P.O. R50PAB85821601, Final Study Report - Part I, Trade Study Results and Conceptual Design Data", LTV data submitted to Rocketdyne, 14 and 15 October, 1985.

8) Personal Communication, S. Simons (NASA-LeRC) to E. C. Hylin (Rocketdyne), concerning failure rates of Grumman dual slot heat pipes for the CBC radiator, 8 May 1985.

9) Briefing Booklet from NASA-LeRC Reliability Working Group Coordination Briefing, prepared by P. M. Finnegan, 23 January 1985.

TABLE 7.8-1

COMPARISON GROUNDRULES

EQUIVALENT BASES WERE USED FOR COMPARISON

* IDENTICAL SDS UNIT SIZE (25 kWe NET)
* IDENTICAL CYCLE STATE POINTS (60 F CIT etc.)
* IDENTICAL SINK CHARACTERISTICS (-50 F)
* IDENTICAL EMISSIVE COATING PROPERTIES (SILVER TEFLON, = 0.816)
* IDENTICAL DRAG/REBOOST SCENARIO ($1145/M^2-YR)
* IDENTICAL STATION BUILDUP SCENARIO (75 kWe -- 300 kWe)
* IDENTICAL EVA AND IVA AND MRMS RATES ($103K, $15K, $0)

* PUMPED LOOP ADJUSTED TO REFLECT PUMPING "HOUSE LOAD"

TABLE 7.8-2

STATION GROWTH SCENARIO

YEAR (IOC +)	SOLAR DYNAMIC POWER kW	SOLAR DYNAMIC CUM. ENERGY (kW YEARS)
0	50	0
3	125	150
7	200	650
10	275	1250
30	275	6750

TABLE 7.8-3

PUMPED-LOOP RADIATOR PARAMETERS

Design Power	90.0 kW	
Panel Width	2.50m	8.19 ft
Panel Length	6.10m	20 ft
Number of Panels	10	
Coolant Fluid	FC75	
Pump Flow Rate	0.351 l/s	44.7 ft^3/h
Pump Power	576 W	

ORU	Mass (kg)	Design Life (y)	Effective Failure Rate (h^{-1})
Panel Set (replaced as unit)	971	20	2.9 E-6
Coolant Management Unit	59	20	4.6 E-6
Connecting Line Pair	13	30	3.0 E-7

TABLE 7.8-4

HEAT-PIPE RADIATOR PARAMETERS

Design Power	88.3 kW	
Panel Width	0.30 m	0.99 ft
Panel Length	7.62 m	25 ft
Number of Panels	51	
Heat-Pipe Fluid	Methanol	
Heat-Pipe Containment	Stainless Steel	

ORU	Mass (kg)	Design Life (yr)	Effective Failure Rate (h^{-1})
HX Boom	406*	20	0
Heat-Pipe Panels (replaced individually)	21 ea.	20	1.62 E-6

*Not including allowance for gas cooler

TABLE 7.8-5

QUANTITATIVE RADIATOR COMPARISONS

Quantity	Pumped-Loop Radiator	Heat-Pipe Radiator
Life Cycle Cost	$570 Million	$798 Million
IOC Cost	$68.1 Million	$76.9 Million
Mass (each)	1,044 kg	1,496 kg
(on growth station)	12,525 kg	16,456 kg
Area (each)	152 m^2	118 m^2
(on growth station)	1674 m^2	1293 m^2
EVA Maintenance Time	77h (154 man/hr)	817h (1634 man/hr)
Module Down-Time	521h	817h
ORU Failures	18.5*	195.6*
MRMS Maintenance Time	10.63h	471h

*Heat-Pipe Failures are much less severe than Pumped-Loop ORU Failures

TABLE 7.8-6

QUALITATIVE RADIATOR COMPARISONS

<u>Quality</u>		
Number of ORUs	3	1* (Heat-Pipe Panels)
Number of Unique Part Sets	More	Fewer
Development Risk	Lesser	Greater
CBC Cycle Match	Better	Worse
CBC Puncture Risk	Lesser	Greater
Ability to Cool Power Electronics	Greater	Lesser
STS Package Volume	Smaller	Larger
STS Package Mass	Lower	Higher

*HX Boom is replaced with PCU and is not a radiator ORU

7.9 ORC RADIATOR COMMONALITY TRADE STUDY

Commonality options for the ORC radiator with the central radiators were evaluated. The trade study is preliminary in nature and is dependent upon more definition of detailed design requirements, cost, and test and verification plans. Initial results indicate:

- The radiator panel design should use common technology but should be optimized for the higher heat capacity and higher temperature application.
- Use an identical contact heat exchanger
- Incorporate a fluid charge (excess heat pipe length for low temperature startup) for the solar dynamic heat pipes only and not on the station.

Table 7.9-1 is a list of all space station radiators with their major characteristics of type, heat rejection temperature, and heat load. The central radiators are closest in requirements to the ORC radiator. The Space Erectable Radiator System (SERS) advanced development program data was used as the basis for the central radiator design comparison. Figure 7.9-1 and Table 7.9-2 contain data appropriate to the SERS configuration. The radiator elements evaluated for commonality are: 1) the heat pipe cross section for both the condenser and evaporator sections, 2) the contact heat exchanger interface, and 3) the heat pipe fluid charge (required for low temperature startup).

The following three options represent varying degrees of radiator commonality and is based upon constructible heat pipe technology.

Option 1: Use identical radiator hardware (ie. same part number) as the central radiator. Lower the ORC heat rejection temperature to remain within the heat pipe capacity.

Option 2: Use the same central radiator heat pipe design (ie. cross section) but with shortened heat pipes to obtain the design heat capacity with the higher temperature application.

TABLE 7.9-1

SPACE STATION RADIATOR SYSTEMS

SYSTEM	TYPE RADIATOR	HEAT REJECTION TEMPERATURE (^{0}F)	HEAT LOAD (kW)	WORK PACKAGE
MODULE BODY MOUNTED RADIATOR	REPLACEABLE THROUGH FLUID SYSTEM DISCONNECTS	35	8 kW	1
LOW TEMP BODY MOUNTED RADIATOR	REPLACEABLE THROUGH DISCONNECTS	-20	.5 kW	1
CENTRAL RADIATORS	CONSTRUCTIBLE	35 AND 70	70 kW TOTAL	2
PLATFORM ESS, PMAD AND EXPERIMENT	CONSTRUCTIBLE	35 AND 70	12 kW	3
STATION ESS AND PMAD	CONSTRUCTIBLE	41	9.5 kW	4
STATION SOLAR DYNAMIC	CBC-PUMPED LIQUID OR CONSTRUCTIBLE	273-60	134.8 kW	4
	ORC-CONSTRUCTABLE	150	230.0 kW	

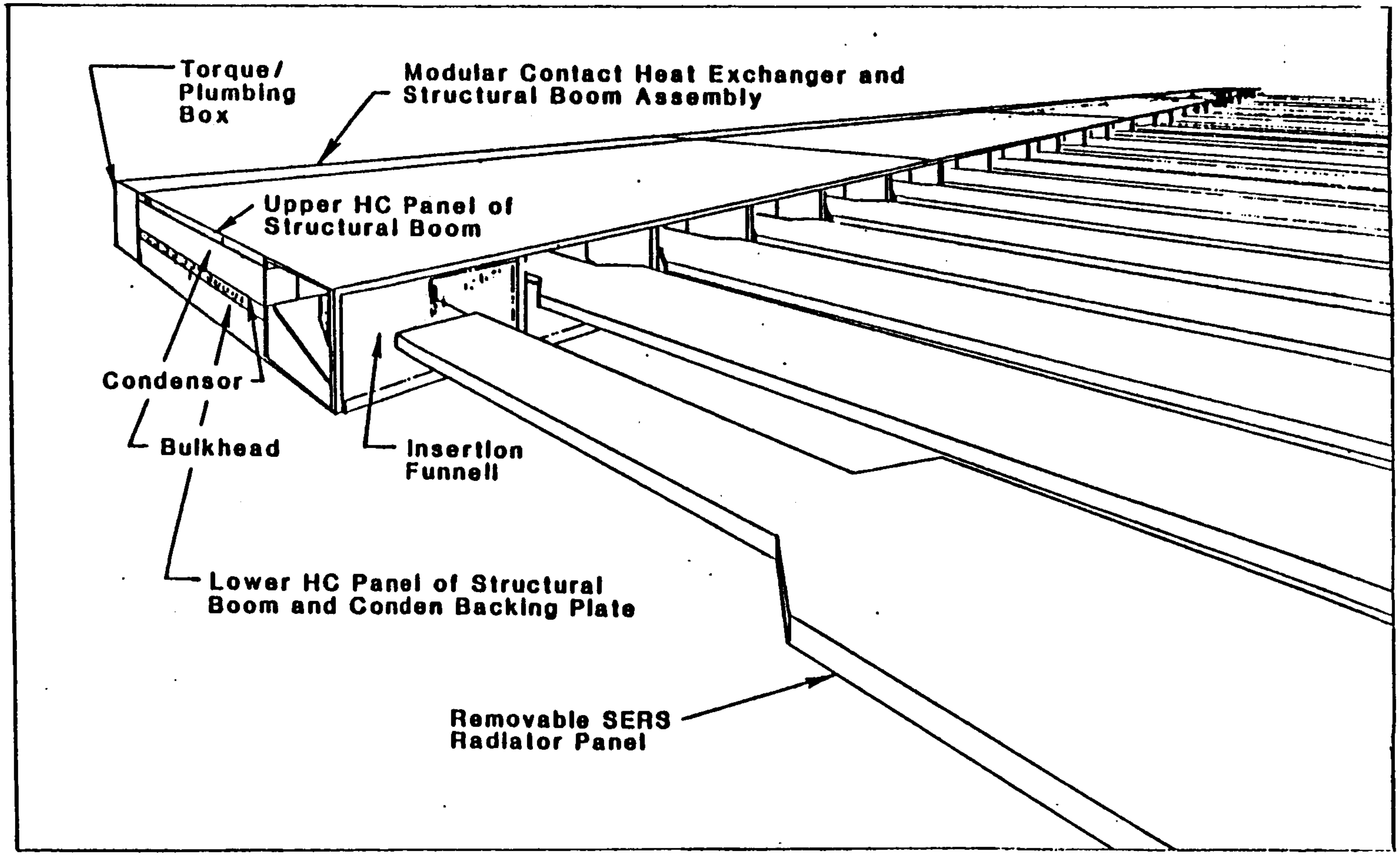


Figure 7.9-1

Edge Mount SERS Central Radiator Concept

TABLE 7.9-2

SERS FLIGHT DESIGN CHARACTERISTICS

- o RADIATOR PANEL
  - LENGTH (TOTAL PANEL): 50 FEET
  - CONDENSER WIDTH: 12 INCHES
  - EVAPORATOR CONFIGURATION: FLAT CONTACT INTERFACE
  - EVAPORATOR WIDTH: 6 INCHES
  - EVAPORATOR LENGTH: 27.5 INCHES
  - CONDENSER LENGTH: 46.6 FEET
  - PANEL THICKNESS: 1.050 INCHES
  - HEAT PIPE: LOCKHEED TAPERED ARTERY
  - HEAT PIPE CONFIGURATION: 6 EVAPORATOR LEGS AND TWO CONDENSER LEGS
  - PANEL WEIGHT: 87 POUNDS
  - HEAT REJECTION PER PANEL @ MAX LOAD: 1.5 TO 1.76 KW
  - HEAT PIPE EVAPORATOR FLUX AT WORST CASE: 11 WATTS PER INCH
  - RADIATION FIN EFFECTIVENESS: 0.9
  - RADIATOR FIN CONSTRUCTION: HONEYCOMB FIN WITH 0.01 FACE SHEET AND 3.1 LB/FT^3 HONEYCOMB

- o CONTACT HEAT EXCHANGER DESIGN INTERFACE
  - INTERFACE DESCRIPTION: FLAT CONTACT HEAT EXCHANGER
  - CONTACT PRESSURIZING METHOD: PRESSURIZED TITANIUM BELLOWS WITH NITROGEN GAS PRESSURE
  - MODULARITY: SIX RADIATOR PANELS PER HEAT EXCHANGER ASSEMBLY FOR CONDENSER; TWO PANELS PER SUBCOOLER MODULE

TABLE 7.9-2

SERS FLIGHT DESIGN CHARACTERISTICS (CONT.)

- CONTACT AREA: 6 INCHES BY 27.5 INCHES
- APPROX INTERFACE WEIGHT: 21.5 POUNDS PER PANEL (NOT INCLUDING CONDENSER)
- CONTACT PRESSURE: 200 PSI
- PROJECTED CONTACT CONDUCTANCE: 1000 BTU/HR-FT^2-^{o}F
- CONTACT TEMPERATURE DROP: 5^{o}F

o SYSTEM CHARACTERISTICS
  - NUMBER OF 70^{o}F PANELS: 33*
  - NUMBER OF 35^{o}F PANELS: 20*
  - NUMBER OF 70^{o}F SUBCOOLING PANELS: 6
  - NUMBER OF 35^{o}F SUBCOOLING PANELS: 4
  - AMOUNT OF SUBCOOLING: 20^{o}F
  - SYSTEM: 6186 POUNDS
  - EST IOC HARDWARE COST: $32.5 MILLION
  - EST TOTAL LIFE CYCLE COST: $73.5 MILLION (NOT COUNTING MAINTENANCE)

* INCLUDES ONE EXTRA PANEL FOR RELIABILITY PURPOSES

** DOES NOT INCLUDE SUBCOATING

Option 3: Use the same central radiator heat pipe technology (ie. same fluids, materials, and methods for panel construction, wicking, and vapor flow passages but with all dimensions optimized for the particular application) and optimize the panel design for the higher temperature application.

All three options assume an identical contact heat exchanger. A comparative cost trade was made. The cost formula is as follows:

Cost = Recurring cost per panel + panel cost per unit area + support boom cost per ft length + cost/unit area due to drag at $567/ft^2 + assembly cost per panel at $30K/panel (15 min EVA) + launch cost at $3200/lbm

The comparative cost includes all IOC costs including DDT&E and production hardware. No reduction of DDT&E cost due to sharing the DDT&E costs with WP02 is reflected in the cost total. Costs for the growth configuration and maintenance were not considered. Table 7.9-3 summarizes the results. Option 3 using the same heat pipe technology but sized for higher heat capacity and optimized for higher temperatures is the least cost. If $4.8 million or $10.8 million could be saved in DDT&E costs due to commonality, then option 3 or 1, respectively, should be considered. These costs must be evaluated at the system level and require more detailed test and verification requirements than are available at this time for an accurate cost comparison. For purposes of this DR02, the reference configuration assumes option 3 and does not consider any cost savings in DDT&E due to commonality.

Heat pipe radiators will not start if the fluid charge is below the artery level due to thermal contraction under low temperature conditions as exist during startup. Two solutions are available. The first is by warming up the panel by rotating it face on to the sun prior to start up. The second method is for charging the heat pipe with excess fluid. For the central radiators, JSC currently proposes the first method. This method requires special pointing and tracking strategy for solar dynamic (the receiver must also be preheated). Solar dynamic proposes the second option. It requires adding a length of heat pipe at the end of the radiator to contain excess liquid at high temperature. Figure 7.9-2 is a graph of condenser heat pipe length needed as a function of

TABLE 7.9-3

COMMONALITY CONCEPT COMPARISONS

CONCEPT	COMMON ELEMENTS WITH CENTRAL SYSTEM	RADIATOR LENGTH (FT)	RADIATOR TEMP (^{o}F)	NUMBER OF PANELS	RADIATOR AREA (FT^2)	RADIATOR MASS (LBM)	COMPARATIVE COST ($M)
1	ALL (ASSUMING CONDENSER DESIGN WILL FIT)	50	96.3	55	2564	6420	56.8
2	ALL EXCEPT SHORTER PANEL	37.8	128.3	56	1929	5371	50.8
3	HEAT PIPE CONCEPT EVAPORATOR INTERFACE	47.3	128.3	31	2273	5046	46.0

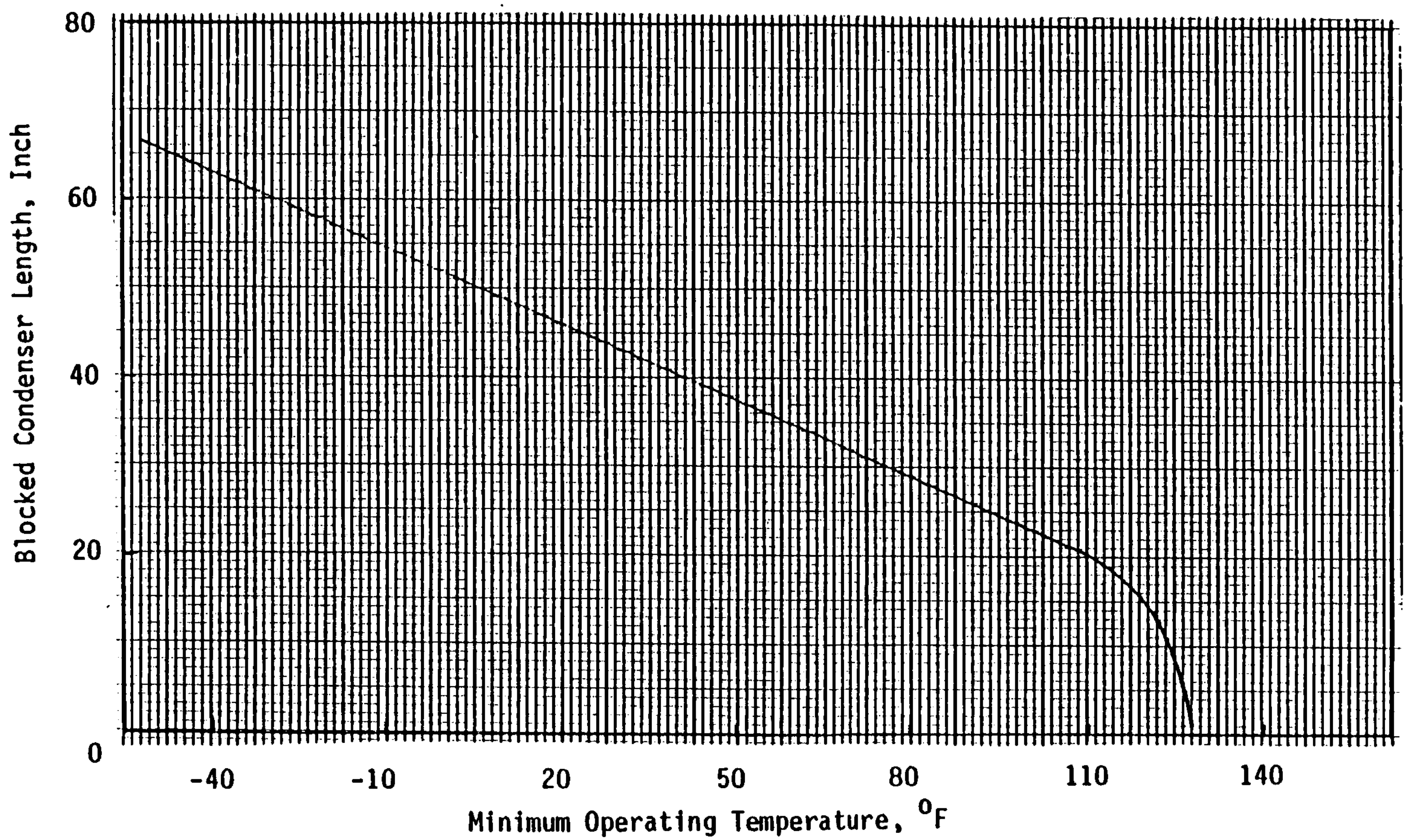


Figure 7.9-2
Effect of Liquid Blockage (128°F) As A Function Of Minimum Heat Pipe Operational Temperature

minimum operating temperature. This heat pipe length could be incorporated as a U-bend within the panel construction at a mass penalty of 3 Kg/m (2 lb/ft) of length needed for each panel. For commonality, the same solution must be adopted for both applications. This area of commonality does not appear at this time to be cost effective.

7.10 Radiator Coatings

A trade study of radiator coatings has resulted in the selection of Z93 white paint as the baseline radiator coating for both CBC and ORC. Silver Teflon is a backup option.

A number of coatings were evaluated for use on both the ORC and CBC radiators; however, all the candidates were of two basic types: white paint and metallized tape. The choices were narrowed to silver Teflon tape of the type used on the Shuttle radiators and Z93 white paint used on the Apollo Service Module. The properties of each are given in Table 7.10-1. The Teflon tape has a lower solar absorptivity, but also has a lower thermal emittance. The lower absorptivity serves to reduce the environmental sink temperature and thus increase heat rejection; however, the lower emissivity reduces the emissive power and lowers heat rejection. Experimental evidence has shown that Teflon can be eroded by the atomic oxygen environment on-orbit. This would reduce the thickness and lower thermal emissivity further. The erosion of Teflon is not great, however, and could retain near initial properties for a number of years by starting with sufficient thickness. White paint with inorganic binders is not affected by atomic oxygen. White paint, however, is more susceptible to contamination by thrusters or ground handling and is not easily cleaned as is the Teflon coating. Since the physical advantages of the two candidates were subjective in evaluation, an analysis of the performance differences was conducted to select one over the other. End of life properties for silver Teflon were assumed to be $\alpha = 0.20$ and $\epsilon = 0.76$. For white paint, the corresponding properties are $\alpha = 0.30$ and $\epsilon = 0.90$. The primary mechanism of absorptivity degradation (excluding contamination) is due to solar ultraviolet flux. Since the solar dynamic radiators will be continually positioned to receive no direct insolation during operation, solar flux will be limited to that reflected by other Station components and the earth. This will amount to less than one tenth of the flux experienced by a non-oriented panel. There will be some exposure of the radiators to direct insolation, however, during the relatively short maintenance periods and perhaps during the initial start-up. Thus, it is believed the end of life properties should be applicable to a 30 year life. Significant contamination from Station thrusters could

TABLE 7.10-1

RADIATOR COATING CANDIDATES

Property	Silver Teflon Tape	Zinc Oxide Paint
End of Life		
Absorptivity	0.20	0.30
Emissivity	0.76	0.90
Weight kg/m^2 (lb/ft^2)	0.33 (0.068)	0.20 (0.04)
Maximum Use Temperature ^{0}C(^{0}F)	121 (250)	316 (600)
Susceptibility to Contamination	low, easy to clean	medium, requires reasonable ground handling precautions
Previous Uses	Shuttle radiator Satellites	Apollo Shuttle Satellites

cause more rapid degradation. This contamination environment and effects are not presently known but will be evaluated by interaction with other Station work packages. Reasonable ground handling precautions should prevent prelaunch coating contamination.

To evaluate the coatings, a TRASYS environmental model of the radiator and concentrator was constructed. Form factor and environmental flux data from this model were used to construct a SINDA thermal analysis model of the radiator and concentrator using an identical nodal breakdown. The thermal mass and front-to-back side conductance of the mirror was modeled; however, the radiator panels were input as having zero mass. The calculated temperatures therefore represent the radiator sink temperature variations around the orbit considering the natural environment and radiant interchange between the radiator and concentrator.

The analysis was conducted for orbit beta angles of 0, 52, and -52 degrees. The 0 degree beta angle orbit produced the highest sink temperatures. Plots of the radiator sink temperatures for this orbit are shown in Figure 7.10-1 for silver Teflon and Figure 7.10-2 for Z93 properties. These results indicate a maximum sink temperature of 213 K (-76^{o}F) for silver Teflon and 216K (-71^{o}F) for zinc oxide. For conservatism, the former value was used for the preliminary radiator designs. The thermal emission can be calculated by:

$$Q/A = \epsilon \sigma (T_R^4 - T_S^4)$$

where: Q/A = Heat Rejection/Unit Area

σ = Stephen Boltzman Constant

ϵ = Thermal Emissivity

T_R = Radiator Temperature

T_S = Sink Temperature

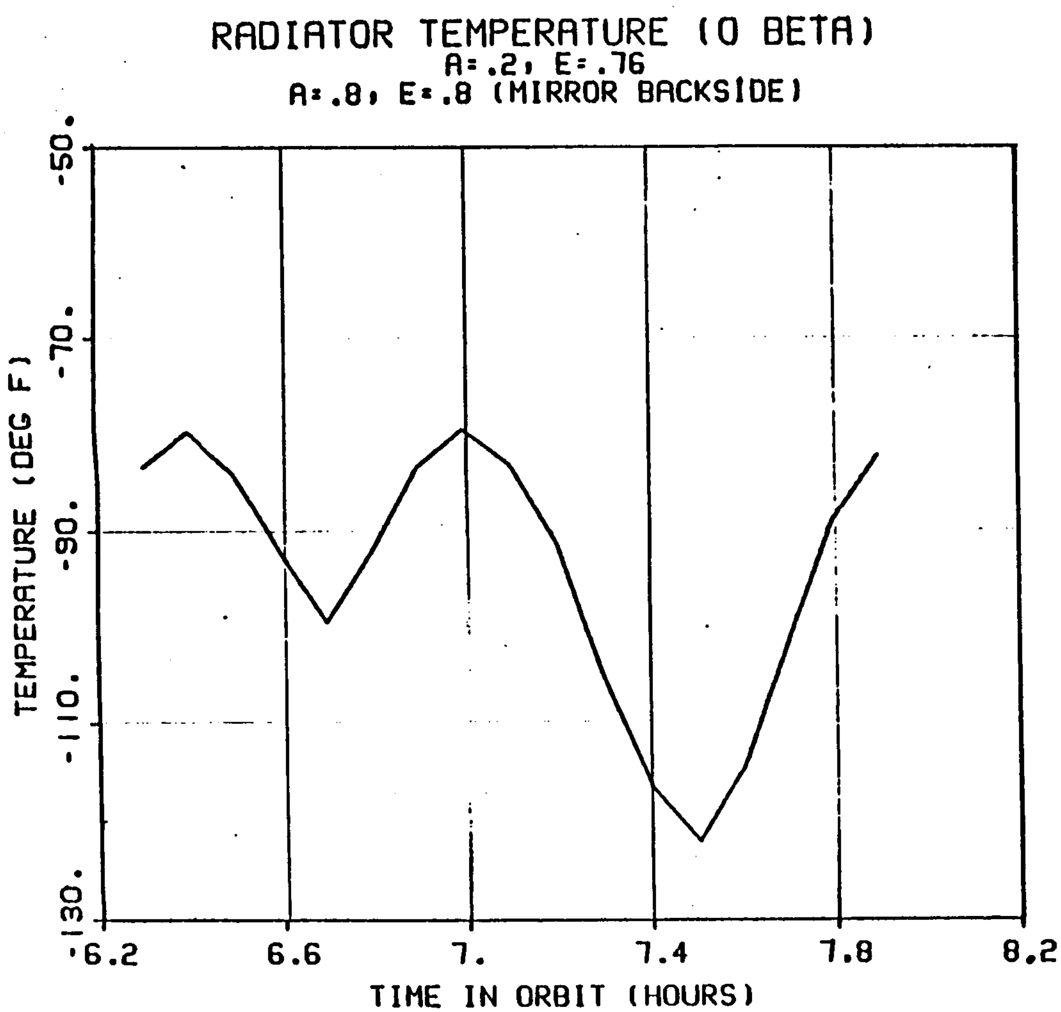


Figure 7.10-1

Calculated Radiator Sink Temperature Using Silver Teflon Coating

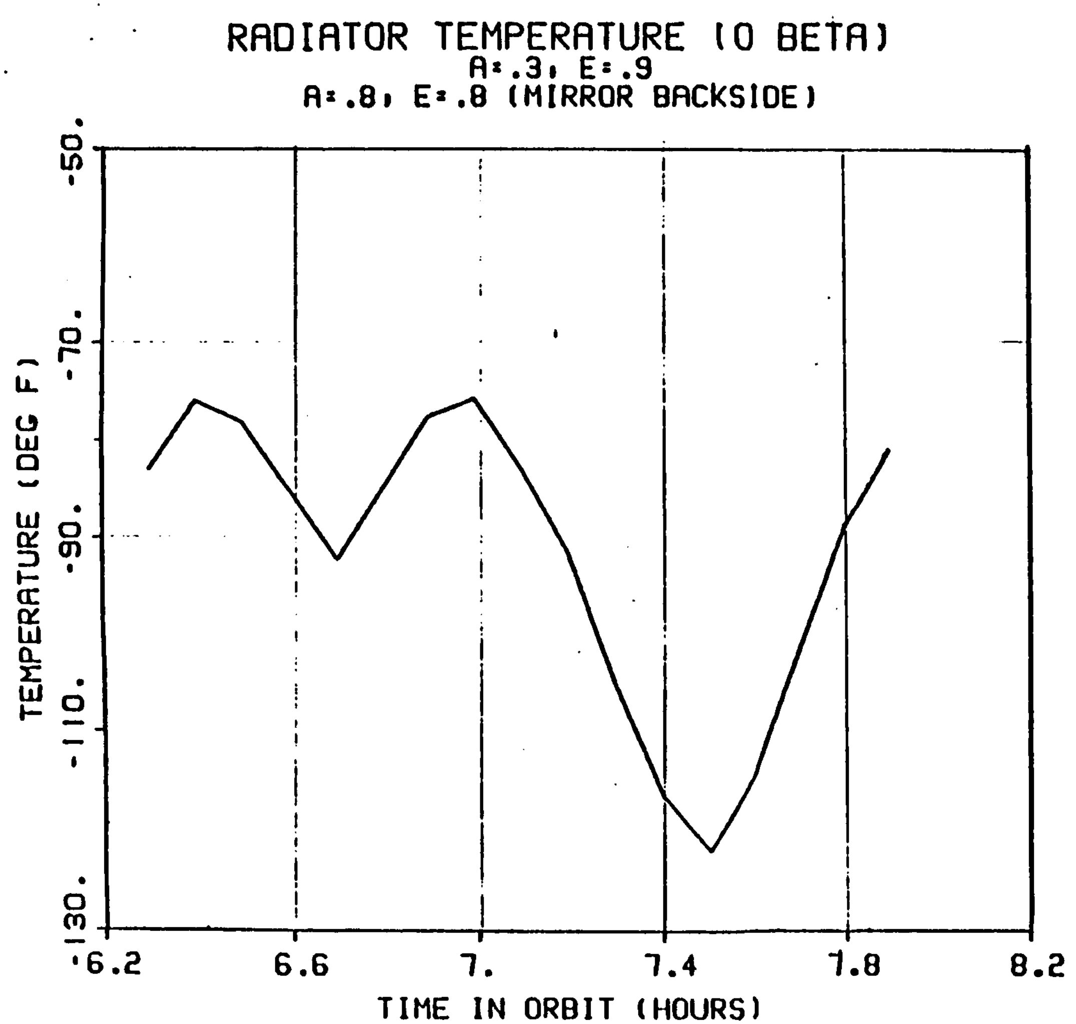


Figure 7.10-2

Calculated Radiator Sink Temperature Using Z93 White Paint Coating

A comparison of these values for a 327 K (130^{o}F) typical SD radiator surface temperature indicate the heat rejection for the Z93 to be 17.5% higher than for silver Teflon. For this reason the Z93 white paint was selected for both the ORC and the CBC radiator coating.

7.11 THERMAL CONTROL FOR SOLAR DYNAMIC ELECTRONICS

This trade study is repeated herein for completeness. The data have been updated and reported as part of the design description in Section 2.2.5.

The SD power module has various electrical components that require some thermal control. Active versus passive cooling options have been evaluated. Passive cooling is the current reference configuration except for the PCU alternator and the PLR rectifier. As more detail has been generated during the Preliminary Design, it has been realized that it may also be necessary to cool actively the PMAD frequency converter that accepts the SD alternator output power. The relatively large heat rejection requirement of this frequency converter cannot be handled by passive cooling. If active cooling is required, the option will probably be to locate the frequency converter outboard of the beta joint.

Table 7.11-1 is a summary of electrical components located within the SD power module and their cooling requirements. The PCU alternator is not shown as it is already cooled by the engine working fluid for ORC and by the FC75 loop for CBC. The total heat rejection requirement is approximately 2.35 kWt during nominal operation.

The frequency converter is 1.5 kWt, or 64% of the total. Currently the frequency converter is located just outboard of the SD beta joint. Four options exist for cooling of the SD frequency converter: 1) Passive cooling, 2) Active cooling provided by the cooling system for the storage batteries, 3) active cooling provided by the SD subsystem, and 4) a separate dedicated cooling loop.

Passive cooling would require a 4.3 m^2 (46 ft^2) cold plate. It does not appear feasible at this time to conduct 1.5 kWt heat energy to the cold plate.

Active cooling by the battery cooling system would require either locating the frequency converter near the batteries or running battery cooling fluid lines to the frequency converter. If located near the batteries, additional PMAD cabling operating at low frequency is required. EMI isolation would need to be incorporated. Running the battery cooling fluid lines to the frequency converter is even less attractive, adding complexity and mass. In either case, growth scarring would need to be considered as each additional frequency converter would require increasing the size of the thermal management.

Table 7.11-1
SD Electronic Component Cooling Requirements

COMPONENT	kWe	kWt***	LOCATION
SD SUBSYSTEM CONTROLLERS			Outboard
Pointing controller	0.20	0.20	
Motor controller	0.30	0.10	
Sunsensor	TBD	TBD	
Insolation meter	TBD	TBD	
PCU controller			
Microprocessor	0.05	0.05	
Other[1]	*	0.25	
PLR controller	0.10	0.10	
PLR			
Nominal	0.0	0.0	
Max insolation	*	0.80**	
Total engine output	*	3.90**	
BETA JOINT			Inboard
Motor controller	TBD	0.10	
PMAD SOURCE			Inboard (outboard if active cooling)
Microprocessor	0.05	0.05	
RBI	TBD	TBD	
Frequency Converter	*	1.50	
TOTAL	*	2.35 +	

* Cycle dependent value
** Based on AC to DC conversion efficiency of 90%, heat rejection dependent on excess engine output over the user load. Values not included in total.
*** Cold plate temperature to be less than 50 C (122 F)
1 Includes motor controllers, analog controllers, etc.

Active cooling by the SD subsystem would require either locating the frequency converter outboard of the SD beta joint or running SD fluid lines across the beta joint. Running fluid lines across a rotating joint is not considered desirable even if the rotation is restricted to less than +/- 180 degrees. The option of locating the SD frequency converter outboard of the SD beta joint could be accomplished with the least amount of increased complexity. Inboard or outboard, power must be transmitted across the beta joint. The SD subsystem already incorporates active cooling for components such as the alternator and the PLR rectifier. and the additional plumbing required to cool the frequency converter would require only a slight increase in complexity. Fluid in the correct temperature range is available. Figures 7.11-1 and 7.11-2 show possible schematics for cooling the frequency converter. In both cases, the radiators would need to be increased in size; 1.5% for ORC and 2% for CBC.

Another option is to provide a separate cooling loop for the frequency converter. This could be provided by a capillary pumped loop and, its own heat pipe radiator located in the same plane as the battery cooling radiator. At least two heat pipes would be needed for reliability. At this time, this appears to be a more complex design approach than SD active cooling. However, more detailed costing is needed for this trade.

The PLR design is based on a direct current resistance radiator. As such, an AC to DC diode rectifier is incorporated in the PLR. This has a relatively poor conversion efficiency. As the PLR normally operates at variable levels at the low end of its operating range and only when there is excess power, the temperature of the diodes could be allowed to rise during infrequent high load transient conditions as long as a reasonable life can be maintained. However, as the design matures some type of active cooling may be required similar to the SD frequency converter. Active cooling for the PLR, however, will not increase radiator area as a less efficient engine is acceptable under these conditions.

Passive cooling is still recommended for the balance of the electronic components. All of the cooling requirements are small and can be accomplished with simple conduction to a cold plate.

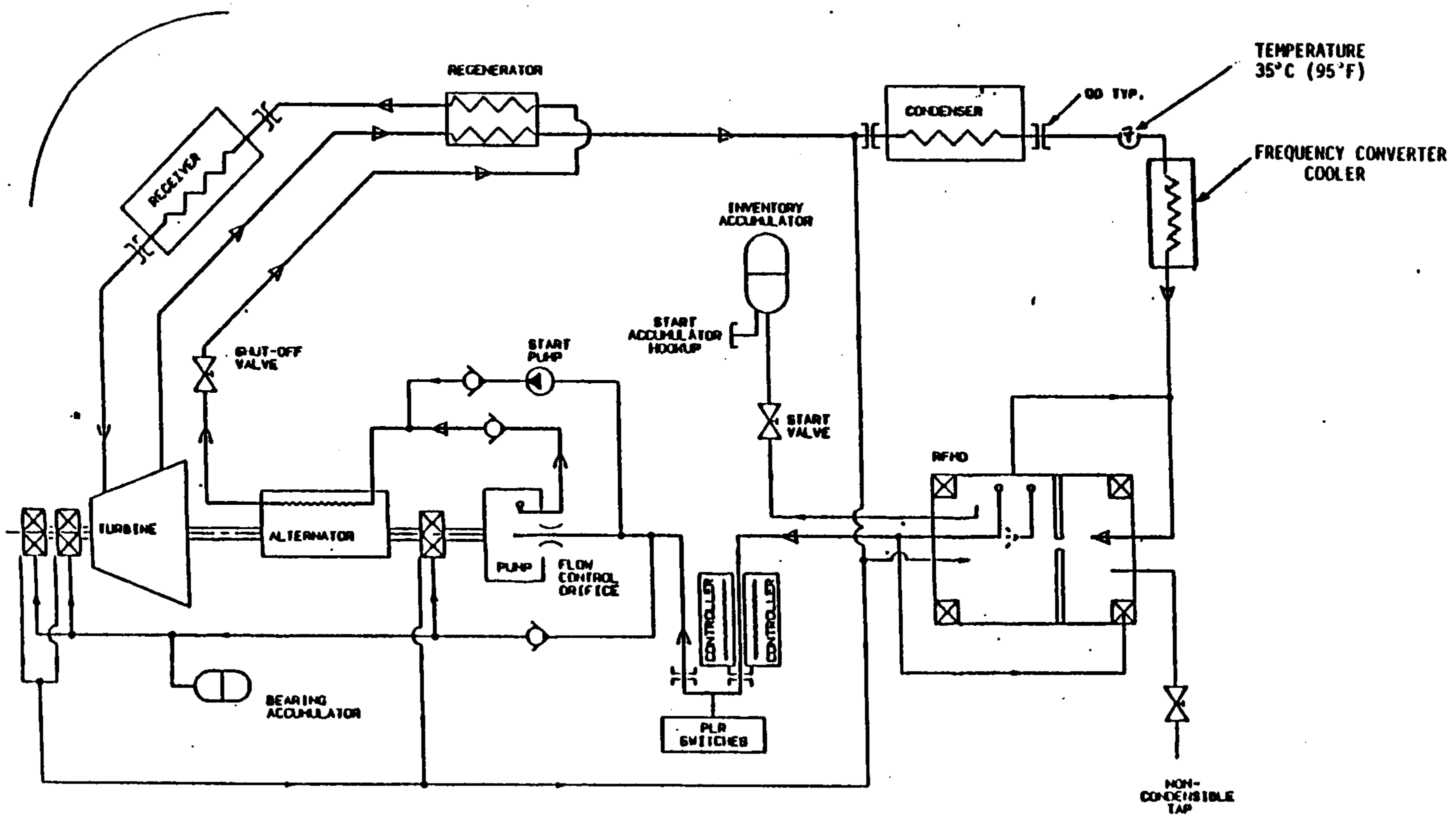


Figure 7.11-1
ORC Schematic For Frequency Converter Cooler

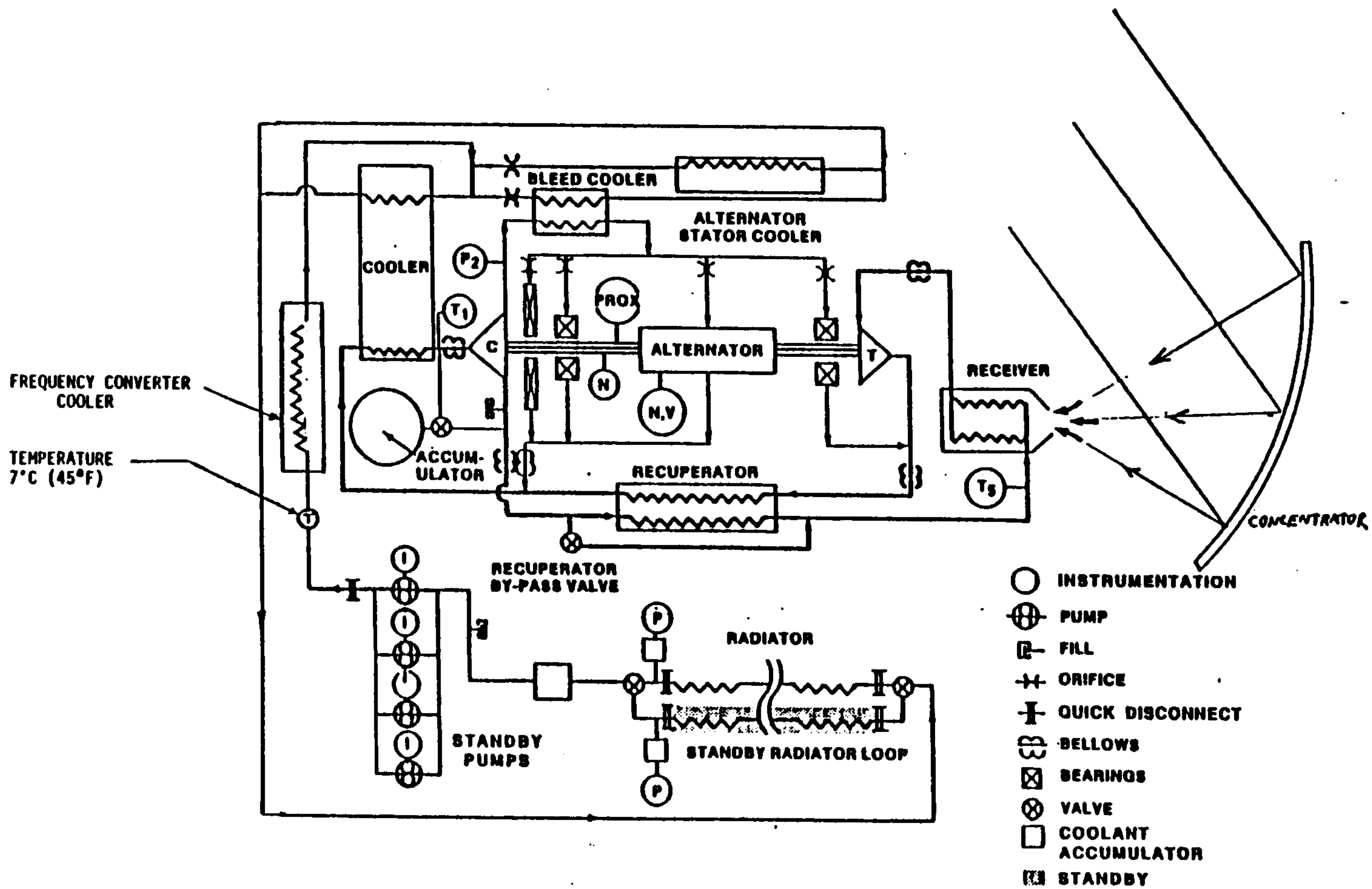


Figure 7.11-2

CBC Schematic For Frequency Converter Cooler

7.12 PMAD COMPUTER FAULT TOLERANCE AND REDUNDANCY

7.12.1 Introduction

The Power Management and Distribution Subsystem (PMAD) computers and controllers will all contain redundancy and fault tolerance to acheive the requisite reliability and availability to accomplish the Space Station mission. Even though the "Standard Computer" ultimately will be chosen by the Work Package 02 contractor, some basic recommendations for features of this computer, particularly with respect to fault tolerance and redundancy management and control appear to be in order at this stage of Contract Development. There are several means of accomplishing desensitization to single-event upsets and other forms of radiation induced faults as well as "hard failures"; however, they tend to group into just a few mechanisms which will be examined for suitability for the PMAD Subsystem on the Space Station.

Perhaps one of the more important issues is the basic architecture of the of the redundancy of the space qualified computer/controller selected. The two most common varieties are dual computer-with or without shared memory using standby redundancy, and self-checking pairs using active parallel redundancy.

7.12.2 Dual Computer-Shared Memory Architecture

In the earlier days of the space program, computers and memories were both relatively expensive. The A/B computer with common memory was the one used most often as the on-board computer for cost, reliability and power considerations consistant with the technology at that time.

In this architecture an active "A" string containing a separately powered CPU,I/O and sensors as well as a separately powered standby "B" string shared a common memory. Inputs and outputs were each cross-strapped so that either computer could input from a sensor or output to an actuator. Only one string at a time was powered. The memory was powered (by the use of OR diodes) if either string was powered. The memory also had a separate "keep-alive" power source (if it was volatile) which was a low-drain battery or alternate power source. The active computer performed programmed Built-In Test which sensed arithmetic or architectural failure (e.g., illegal instruction) and sent a pulse to a watch-dog timer typically once per second if the B.I.T. was successful. The watch-dog timer typically was set to expire if it failed to receive an "I'm OK" pulse within 3 seconds (indicating either a computer failure or a software hang-up.) Expiration of the timer would remove power from the powered string (except the memory which used its keep-alive power source (if it was volatile) and generated a signal for operator intervention. The operator could then either repower and reinitialize the A string or switch to the B string by powering the B string. (For a space mission the operator was usually ground control.) The use of the operator or ground control to switch over to the backup processor resulted from a reluctance to "trust" the computer compared to a human.

The shared memory type of redundancy usually employed 6 bit EDAC (Error Detection and Correction) code to detect memory errors with a higher degree of confidence than the single bit parity method and to correct single-event upsets or single errors without reinitialization since there was no program storage redundancy except by repetition of the code in memory or the use of ROM if provided. Since power is removed from the active computer upon detection of an error by this method, restoration of power or application of power to the other string first would have to cause the memory contents to be checked for possible damage/alteration (which may have been caused by the erratic computer); and to either correct stored errors by the error correction mechanism or reload memory from the ground or on-board mass storage devices when required. The effects of the erratic computer on memory contents virtually can be eliminated by incorporating memory protection which is usually used with this architecture.

The memory check may cause interference with the on-going mission by denying the computer to it until the memory is filled and checked and the computer reinitialized. This type of interference usually is not important to a satellite with primarily navigational mission; one in which denial of the reaction jets or sensor data for a few seconds, should not cause excessive satellite positional drift and usually can be corrected a few seconds later with no serious loss of data or accuracy.

These earlier space on-board computers such as the Litton 4516E were special hardened/highly reliable implementations of the then state-of-the-art in militarized computers and were comprised of predominantly MSI and/or IC "S-level" parts and were relatively heavy, consumed relatively high power and were fairly expensive (about $1M each). For reasons of cost, weight, reliability, power consumption and mission type, most space computers used a dual computer string (one active, one standby) with shared memory at that time. In the present era, single-chip qualified computers already do or are about to exist wherein computer and memory cost will be substantially lower as will weight and power.

Missions which are more sensitive to momentary computer "glitches" will be more common and missions which can "coast" through a dropout will be rarer. All of the above will tend to favor architectures which will use equipment more liberally in implementing redundancy.

7.12.3 Self-Checking Pairs Architecture

Foremost among these architectures is the self-checking pairs implementation, which is one wherein each processor is a synchronized pair and is comprised of two relatively cheap one or more chip CPU,IOU, and memories. Certain registers, typically the memory data and memory address registers are compared each memory cycle. If they compare favorably, it is presumed that both of the computers in the pair are OK. If they are different, it is assumed that one or the other computers of the pair has failed and therfore that the computer pair (the A pair) is not suitable for use. The redundancy within the pair is used only as a means of determining failure without resorting to software tests. For mission continuance, it is necessary that a second set of computers the (B pair) ,at least, be provided to continue the mission. Note that the second set also must have been running from time zero so that its memory will have current data at the time of failure. Once a set has "failed", there is no known easy proven practical way at the present to salvage or reinstate the remaining good computer/s (if there are any) during flight to continue the mission except by physically or electronically replacing the pair. The number of self-checking pairs required must be determined based on mission success requirements. In-orbit module repair/relacement by EVA or other means (robotics) will improve availability. Another important issue is the degree of confidence in the ability of the computer to detect the fact that a fault has occurred; and for certain missions, the speed with which the fault is detected.

The self-checking pairs implementation has a lower (worse) mean time-to-failure (MTTF) than a single computer of the same species due to the duplicate hardware and the comparison circuits. As a result of the lower MTTF, consideration to recycling the "failed" computer due to single-event upsets is of increasing importance.

7.12.4 Other Experience

Of course, voting configurations can be used in which three or more computers are used and if two or more are in agreement, they are considered to be correct. The two-out-of-three voting triad configuration solves some of the concerns with the self-checking pairs implementation while retaining all of the advantages. For example, it is known which computer of the triad is the non-conformer and operation can continue with the remaining two in parallel redundancy or some other configuration ;thereby, eliminating the need for yet another set of processors and extending useful life to two failures rather than the single failure permitted by self-checking pairs without recycling "failed units". This results in a 2.64:1 improvement in MTTF. A three-out-of-four configuration has about three times better MTTF than a self-checking pair. It can be seen that increasing the number of computers in voting architecture beyond three has only limited effect which would be almost completely offset by the lower reliability of a unit computer due to the added comparison logic.

There have been some missions/uses where cost was not the paramount decision factor such as the SSME (Space Shuttle Main Engine) Block 2 where life safety considerations were the key factor. This program used an architecture which was a hybrid of the two major types discussed above and assigned a separate memory to each computer and because of its required response time of about 20 ms., it performed A/B switchover automatically. It used self-checking pairs because of the need to detect a failure with a very high confidence level. Due to the short active mission of about ten minutes only one extra pair with no means of reactivation was provided.

The Global Positioning Satellite's Digital Computer Equipment Assembly (DCEA), for example, uses an Autonetics A/B computer with duplicate data memory very similar in concept to the Litton computer except that it employed a separate shared program ROM to facilitate restart.

Modern missions may no longer be able to rely on humans as the primary means to switchover to the backup equipment because rapid response times required for certain mission type continuance may be faster than possible by real-time man-machine interface.

7.12.5 Practical Considerations

It develops that the practical aspects probably will be more important than those derived by theory or logic in determining desired characteristics. For example:

1. The difference in mission reliability between standby redundancy and active parallel redundancy (self-checking pairs) differs by less than an order of magnitude. More redundancy or the use of in-orbit repair/relacement is needed to effect a significant improvement. This suggests that the basic form of the architecture and redundancy should be selected for other more practical reasons.

2. The confidence with which a fault is detected may be the most important factor and strongly points to the self-checking pairs or two-out-of-three voting triad approach regardless of all other factors.

3. The self-checking pair or triad does not distinguish between hard faults and single-event upsets which means that for long missions the

number of single-event upsets (typically about one a year) could equal or exceed any practical number of redundant units provided unless some means of recycling "faulty units" is included. It should be noted that units which "failed" due to single-event upsets are probably still good after being reset (to a fairly high confidence).

4. Automatic switchover after failure (or upset) is probably necessary because some of the loads are life critical or can not be interrupted for more than a few milliseconds such as the Solar Dynamic Controller. . One might consider the use of two different approaches; one for life critical conditions and/or rapid response, and a separate one for non-life critical or non-rapid response.

7.12.6 Preliminary Recommendations For The Space Station

1. The use of single-board or single I.C. self-checking pairs or two-out-of-three voting triads. (for high confidence in detecting faults).

2. The use of at least two active redundant sets of the self-checking pairs or two-out-of-three voting triad setsfor mission availability after a failure.

3. Automatic switchover in the case of failure- with manual/ground control intervention permissable.(for rapidly continuing operations for single-event upsets in non-memory area)

4. Use of at least dual sensor inputs and outputs to avoid single-point failures.

5. Use of duplicate EDAC memory for each self-checking pair/triad to dispose of single-event upsets before they present problems (the memory is approximately ten times more likely than the CPU to suffer single-event type upsets due to its larger area. (The number of hard failures can impact long missions; particularly a 10-20 year mission).

6. Eliminate all single point failure mechanisms.

7. Provide for in-orbit repair/replacement to reduce the number of on-line spares and logistical spares requirements.

8. Use PROM backed up by the central DMS and the ground for program backup storage.

9. Use a computer having good Memory Protect features.

10. Use a watch-dog timer (to detect software failures/hangups).

11. Use dual power supplies each fed from a separate power source/bus to reduce single-point failures.

Further study using hypothetical faults and system requirements is recommended on at least the following:

1. Redundancy of the self-checking (comparison) circuits.

2. Redundancy of the sensors.

3. Architecture of the switchover mechanism.

4. Hardening level and type required.

5. Which registers are best to use for the self-checking.

7.12.7 Sample Scenarios and Other Considerations

Sample Scenario with Self-Checking Pairs

Assume a self-checking pairs implementation with separate non-EDAC memories and that a processor such as the PDCU sustains a single-event memory upset by EMI or a high energy particle such as a cosmic ray which causes only a momentary failure:

1. The self-checking pair 'A' senses a discrepancy between its primary processor and its monitor processor.

2. Having no direct means of distinguishing between a hard failure and a single-event upset,the PDCU is switched to the 'B' set of processors by the comparison hardware and bits are posted in the health monitoring and redundancy management/control computer (the Power Management Processor)) indicating the changed configuration.

3. At a later time; preferably when the activity of the PDCU should be stable for a while , or at a prearranged time-slot,one or more self-test programs are run on the "failed" computer. If it shows no error for,say, three times in a row,it is declared to be good and is restored to active parallel redundancy by some means such as perhaps, transferring certain memory contents from the now active computer then transferring the current insruction address to the address counter/register of the previously "failed" computer. Bits are posted in the A/B control logic indicating that this computer is now the B computer. Care must be taken not to allow changes to memory of the active 'A' computer while the newly restored 'B' computer is being reinitialized, etc. This system may be difficult to implement successfully. Other means/rationale must be developed to restore a previously "failed" computer such as the use of a double buffer in the software for inputs and outputs. If the computer tests bad for n-out-of-m times, the computer could be declared a candidate for replacement by means of EVA or robotics at a later time.

If the scenario had included EDAC memories, the single-event upset would have showed up as a "failure" in the memory which sustained the failure only. It would have been corrected immediately by the EDAC and would not even trigger the self-checking pair comparator alarm. It would be desireable to send a signal to the health monitoring computer (the Power Management Processor)indicating that a correctable failure had occurred in computer xx at time yy. A trend analysis could be used to, e.g. count failures over a time period. If the trend surpasses a threshold, it could declare the computer to be questionable (at least) and perhaps also alert the operator/ground which might ask for more extensive tests or could switch to a different computer pair.

Sample Scenario with Two-out-of-three Voting Triad

Assume a scenario with a two-out-of-three voting triad where each processor of the triad has its own non-EDAC memory and one of the thee processors sustains a failure.

Processor #1 has the failure and is detected as being different than Proceessors #2 and#3. Computation continues on Processors #2 and #3 as a self-checking pair and Processor #1 is deactivated,cleared and initialized. Then a self-test program is run on Processor #1 to determine if this was a hard failure. If Processor #1 passes the test,say, three times in a row, it can be presumed that it was a soft failure/single-event upset. We now are back to the problem of placing a "recycled processor back on-line only with three processors it is really no easier.

Recycling a "Failed" Computer

Each computer program must be designed to permit recycling of "failed" computers. A specific time-slot must be provided wherein the active program can idle while the "failed" computer is initialized and placed on-line. Any real long-term data will have to be stored in at least two places to prevent/minimize its chances of being lost due to a failure. Short-term data should be designed so that it can be reconstituted rapidly as new inputs are received.

Typical PMAD Processor with 1553 I/O

A typical PMAD processor as shown in Figure x consists of an A and a B redundant channel of self-checking pairs or 2-out-of 3 triads ; each processor with a separate EDAC memory and having cross-strapped inputs and outputs so that any processor can access any input or drive any output. The balance of this paragraph will assume the use of self-checking pairs to develop I/O concepts. The 2-out-of 3 triads could also have been used.

There are two dual 1553B I/O channels; one for processor to processor communication and one for driving the RPCs and RBIs. Each 1553 I/O channel interfaces with an MBIU (Multiplex Data Bus Interface Unit) which serves to unload and decouple the processor from bus tasks such as polling, bufferring, interrupt and storage protocols until data is absorbed by the processor's memory by infrequent DMAs from its MBIU. The MBIU receives the bus messages addressed to the processor, synchronizes them to the processors and inputs them to both halves of the processor's self-checking pairs identically. On output,the MBIU is driven by the "good" self-checking pair. Each pair has a dual comparator which compares memory data and address with its twin. If they agree, both are assumed to be good. If they disagree,both are deemed bad and are not used except after further test and reinitialization.

Each channel has a separate power supply supplied by a separate power bus. A watch-dog timer pair is provided for each channel. If no errors are detected by the inherent hardware self-tests in each processor,every time the processor passes a predetermined address, it sends an "I'm OK" pulse to the watch-dog timer,resetting it. If an error is detected the timer expires (outputs immediately) . If the timer expires because it has not received a reset pulse in the period before expiration)due to a software/hardware hangup, the timer fires and triggers control signals which cause the then active channel A or B to switch off and switch to the surviving pair/s. It should be noted that this paragraph dealt with self-checking pairs as a convevience in presentation; two-out-of-three voting could just as well have been used.

7.13 PMAD BUS ALTERNATIVE STUDY

7.13.1 Introduction

The PMAD subsystem control function needs a means of communicating between the processors, micro-processors, controllers and load and power bus control devices such as RPCs and RBIs. The question is should this be by means of a bus, a group of buses, a combination of buses and direct computer I/O, or a common bus with the DMS bus? A further question is should there be more than one set of media for the buses that are required? Also, what should the message standards and protocol be for the bus? This study will address these questions, not necessarily in the above order.

7.13.2 The RPC Interface

The largest number of devices, as a group, is the approximately 1000 RPCs and RBIs needed for load control of the growth configuration. These are housed in groups of approximately 50 in the Power Distribution and Control Units. The wiring distance between the RPCs and their PDCU Processor is quite small and is more a matter of back-plane wiring than of long-haul cabling.

This suggests that something akin to normal computer I/O would be better than a bus because of the simpler interface, lower cost and power and potentially more rapid responses. The choices of interface at this level are basically just serial or parallel (or the use of a higher speed serial interface such as MIL-STD-1553B could be studied).

A parallel interface such as the IEEE 488 bus would require that about 24 wires (the data and control lines) be brought out of the PDCU Processor and go to each RPC and RBI in the PDCU. Since this is back plane wiring, this poses no undue hardship and the cost is reasonable. A direct parallel I/O interface would only take a few microseconds for commanding (actuating) or reading (obtaining status/current value) any one RPC or RBI. Furthermore, any RPC or RBI that wants to talk to its local PDCU processor, can interrupt and get virtually immediate attention without consideration of bus access delays. The only delays would be interrupt and job rescheduling times in the software of the controlling PDCU Processor which should be less than one millisecond. Should another processor such as the neighboring PDCU want to control or interrogate an RBI or RPC, not on its I/O network, it would do this by first accessing the controlling PDCU via their common bus, then have the controlling PDCU in turn, access the RPC or RBI via its software and I/O. Only a single parallel interface card would be needed in each PDCU processor. The IEEE 488 is about twice as slow as parallel I/O, but is still quite fast as a one-way bus. The IEEE 488 protocol permits any user to request service by energizing a common control line to the bus controller. The bus controller must then poll all users on its net to determine which one wants service; then interogate (permit) it. This could reduce polling time requirements by more than a factor of two from the 1553 bus. The use of parallel polling also reduces polling time.

A comparison of power requirements reveals that the 1553B interface may be implemented with a 600 mw hybrid chip, a half-dozen CMOS IC's and one 300 mw line driver for a total of about 1.0 watt per interface group. The 488 interface may be implemented with one 100 mw CMOS I.C. but requires 24 line drivers at about 150 mw each for a total of about 3.6 watts per interface group. One 488 or 1553 controller interface per processor will serve its approximately 50 RPC/RBI loads. Each RPC and RBI would have to incorporate a receive/transmit "slave" terminal capability so that the 1000 RPC/RBI's would dissipate about 3.6 kw in contrast to only about 1 KW for 1553.

If a serial interface such as RS-232 were used, other system costs would increase. For example, a serial interface would require about 30 serial interface cards to accommodate 100 devices (3 RS-232 interfaces per card) and an individual wire bundle of about 4 wires would have to link each device with the PDCU processor. The RS-232 interface would operate at approximately 1200 bits/sec and passing just one 16 bit word would take 13 milliseconds which is slow compared to the parallel I/O or IEEE 488 interface. It is fairly obvious that the parallel I/O or IEEE 488 interface is superior to RS-232 due to the fact that only 1 card (already existing) is required per PDCU processor in contrast to the 30 cards required for the RS-232. The parallel interface would require 24 wires linking each processor with the PDCU whereas the serial RS-232 would require 100 sets of 4 wires; one to each RPC/RBI as interconnects. The IEEE 488 also is much faster. At this point, no further consideration will be given to the use of RS-232. The 1553 bus should be considered for use at the RPC/RBI interface level due to its lesser power and number of wires. The same analysis will set the framework for the potential use of 1553 for PMAD interprocessor communications and is presented below.

7.13.3 The Processor-to-Processor Interface

The PMAD processors, micro-processors and controllers should share a common bus with a common interface, if feasible. Some of the data exchanged, particularly fault data, and Solar Dynamic Engine control data must reach the intended recipient as quickly as possible to reduce the risk and extent of potential damage; and certainly within the permissable over-stress times which are typically 25-50 milliseconds for most power equipment.

Closed-loop control data, such as is used with the Solar Dynamic Engine on its local area bus with its Controller, likewise, should reach the recipient rapidly; the permissible time being a function of the controlled device and its thermal and overload characteristics.

There are about 25 processor/controllers currently anticipated to populate the growth configuration on the PMAD Control bus. The PMAD Control bus would have to support that many subscribers while maintaining acceptable response time. Various networks could be studied to find the one best suited for PMAD use. Due to the desire to share a common network with DMS, particular attention will focus on CSMA/CD type buses which are being contemplated for DMS.

7.13.4 Use of a CSMA/CD Bus

Normal baseband communications networks typically use a bidirectional signal path on which signals are encoded on the cable using Manchester or other baseband techniques. A variety of packet mode access techniques can be used. The most common implementation is CSMA/CD (Carrier Sensor Multiple Access with Collision Detection) in which all subscribers share a channel/wire/fiber.

Subscribers decide to transmit when a channel is free (no carrier). If two or more subscribers decide to transmit at the same time, collisions of data packets will occur. Subscribers then retransmit after a randomized delay.

Once a subscriber starts to transmit, it can be either a short or a long transmission limited only by the network protocol rules to some maximum message length, usually in the order of 256 bytes (2048 bits). If another subscriber needs rapid bus access, it is conceivable, though unlikely, that he may virtually never get access or that access time may be denied for a substantially long period since other subscribers might repeatedly get access before he does. For this reason, rapid-resonse cannot be guaranteed on a CSMA/CD type bus. The PMAD subsystem has some signals and control loops, in which the worst-case response must be known and be less than a prescribed value. a deterministic bus, one in which the worst case response time can be calculated, must be used in this case. A CSMA/CD type of bus is not recommended. It is possible to have collision detection and some form of determanism by using a non-standard protocol. Some of the candidate DMS buses profess this kind of bus; one in which a detected collision caues every subscriber to revert to a different predetermined delay before retransmission. This approximates a fair-share bus. In effect, this is setting priorities. If all or most users are the same priority, this really doesn't help much. Furthermore, it presumes that all subscribers sense the collision which is probably over-optimistic,pareticularly for a fiber-optics netork.

The next most common bus, particularily in MIL-SPEC type equipment is the MIL-STD 1553B.

7.13.5 Use of MIL-STD-1553 Bus

The 1553 bus is a time division command/response multiplex data bus which is more exactly a master/slave point-to-point data link with several users sharing a common media (cable). Not all 1553 buses are directly compatible due to the open definition of certain fields in the MIL-STD. The 1553 bus is usually used because of its relatively high speed and because there are readily-available low-cost IC's which implement the receive/transmit and in some cases the address recognition function. The balance of the protocol and decoding is done in the software of the net controller/s and remote terminals which can either be a portion of one or more existing system processors or can be a dedicated small processor.

Due to the 1 mHz bit rate and the fixed short message length of 16 bits plus parity and sync and the relatively large space between messages, a small buffer can be used, thereby reducing the cost and complexity of the MIL-STD-1553 receive interface device.

The 1553 bus is not suitable for very high speed use but is virtually ideal for use in exchanging short messages as in a low to medium speed power control system.

The first limitation on using 1553 is that the address field can only accommodate 30 slaves/terminals; however the subaddress field can accommodate 30 address variations for a total of 900 possible unique addresses. Since the standard IC's only decode the address field, the subaddress would have to be decoded in the slave processor; thereby possibly slowing it up; or generating a requirement for a more complex interface at the receiving terminal/slave containing at least a one chip logic device/processor in addition to the interface IC, to decode the subaddress.

One other limitation of the 1553 master/slave protocol is that the slave can only talk when told (allowed) to. This imposes a requirement upon the master (the net controller) to poll all the slaves periodically to permit them to send data/status back to the master or to a designated recipient.

In a centralized-processing PMAD system, theoretically, a single 1553 bus could be used to link all PMAD equipment; however due to the large number of RPC's (about 1000 for growth) the polling and addressing requirements would be excessive on the processor used as net controller and the effective respones time of any one RPC slave/terminal also would be excessive.

A practical 1553 system can be used for the distributed processing PMAD subsystem where only the procesors and controllers which number less than 50 for growth are placed on one or more 1553 Control buses and can exchange information with each other by means of the Power management processors acting also as net controllers. Each PDCU would serve its associated RPCs and RBIs over local 1553 networks.

Response Time of the 1553 Bus for the Computer Network

In order to test feasibility to control about 50 Processors and controllers on a single 1553B bus, a quick analysis is presented herein where time to poll and send a message to one processor is estimated:

Action	Instruction	Time
1) pick up next poll address	4	12 usec
2) construct message	12	36 usec
3) message bit transmit time		20 usec
4) Intermessage Delay		12 usec
	total =	80 usec/user

The response time of each receiving processor's status message would be comparable for a total of about 160 usec/user round trip time. For 50 users on one bus, this would be 8000 usec or 8 ms as the maximum time neglecting software time) it would take an interrupt type of condition to reach the net Controller as a result of polling and access delays. This should be sufficiently fast for either PDCU or Power Management processing.

7.13.6 Dedicated or Shared Communications Net Controller

The quick analysis of response time presented above implicitly assumed a dedicated net controller processor since it neglected all other processor tasks. The main concern in the case of a non-dedicated processor is the time it would take for the software job controller in the processor. Even a relatively simple operating system, executive or resident data management system would take in the order of a millisecond every time there is a job change or a switch from I/O to procesesing. It is rather obvious that the polling task could not tolerate such an environment without special considerations such as high task priority, DMA(Direct Memory Access), and a high-speed executive to maintain rapid response time. An interrupt driven I/O processor as part of the net controller processor should also be considered if an existing system processor is used as the bus controller. The safest approach would be to use a small dedicated communication bus controller processor. Such equipments are available both in commercial and military equipments and their use is recommended to unload and uncouple the net controller processor.

7.13.7 Ability of User to Communicate with Bus Controller

One factor of paramount importance in the case of slave processors in a real-time control system is to be able to communicate an emergency condition as soon as possible to the net controller/processor from whence it would be forwarded to the concerned processor/user. As discussed above, the 1553 interface is dependent on polling rate from its bus controller. This either limits its suitability or requires a dedicated polling processor; perhaps a one chip processor. This is recommended.

7.13.8 Use of An IEEE 802.4 Bus

The IEEE 802.4 bus is a representative deterministic, token passing bus using a common media wherein each active subscriber broadcasts when he has the token. All other active subscribers listen whether or not they are interested in the data. When a subscriber has the token, he can either transmit data and then pass the token; or as a minimum just pass the token. A message is defined as a data block of variable length surrounded by a preamble/header and the token. The maximum message length is 8191 bytes or approximately 65k bits. Each active subscriber must get a fair turn as the token is passed around the "logical ring" of active subscribers. A new subscriber can enter the bus at the entry window at the end of the last active subscriber message. an active subscriber can transmit more than one message when it is his turn, but this is usually frowned upon and a priority scheme to permit this has not evolved yet in the standard.

7.13.9 Use of a Common IEEE 802.4 Bus with DMS

If one shares even a deterministic bus of this type with many users over which you have little control it is very easy for your bus access time to escalate momentarily to fairly large values. The following sample calculations will show this:

Assuming a 10 megabit bus:

Worst case message length = 65,000 bits = 6.5ms/msg max. bus
rate 10,000,000 bits/sec

If there are about 130 users of which 50 are in the PMAD system and the rest are comprised of about 40 housekeeping users and 40 payload users. Any one PMAD subsystem user could expect a worst case access time of approximately 100 times the worst case message length or .6 seconds; since he would only get the token about 1/100th of the time if everyone were active and transmitting worst case messages. In a practical situation, everyone will neither be active nor transmitting worst case messages. For example, we know that the PMAD subsystem messages are much shorter (a few hundred bits at most). It is possible, even though not very probable that an occasional response delay of a common bus would be in the order of a half second and could delay PMAD corrective action long enough to cause either unacceptable damage; or short term loss of power to a critical load.

It is also possible that any of the other users could inadvertently hog the bus and cause even longer delays if not properly controlled by either its bus controller's local fault detection and correction or the bus's hog control messages.

There are several other factors which add to the access delays. These are relatively small delays and usually can be neglected compared to the larger delays cited above. These include:

a) Cable delays (propagation time)

b) First time net entry delay

c) Host processor "digestion delays"

d) Bus interface unit buffer delays.

It should be noted that the worst case delay can be limited if the maximum message length is shortened from that in the IEEE standard. This would be all right for PMAD but might be detrimental to other users, particularly those with lots of data to transfer. (e.g. the communication subsystem).

7.13.10 Response of a Dedicated PMAD Network

The response of a non-shared IEEE 802.4 Bus PMAD network would be as follows for the protocol worst case message length:

Worst access:

$$\frac{\text{worst case message length}}{\text{bus data rate}} = \frac{65{,}000 \text{ bits} \times 50 \text{ users}}{10{,}000{,}000 \text{ bits/sec}} = 325 \text{ ms.}$$

This is probably too long; however, it is easy to make shorter by reducing the maximum message length protocol (for PMAD only). If a maximum of 256 bytes were used, the worst case response would improve to 256 x 8 x 325 ms = 10.2 ms. or less, 65000 which is probably acceptable. A maximum message length of 256 bytes is more than acceptable for PMAD since the messages are short and there are no known long messages which require rapid response. In fact, further reduction of the PMAD worst-case message length could be considered if better response time should be necessary.

This analysis shows that a representative IEEE 802.4 bus with protocol restrictions is acceptable as a network since it is deterministic and the worst case access/response is acceptable; however, sharing an IEEE 802.4 bus on a common network with DMS is not acceptable due to the increased traffic and the standard protocol maximum message length. At the time of writing this document, no practical components exist for a 802.4 bus; however, a token passing ring type bus which has similar characteristics should have viable components shortly. Another possibility would be a broad-band bus two channel network which would allowseparate rules for protocol to exist in each of the two networks.

7.13.11 <u>Use of a Broad-Band Bus on the Space Station</u>

A broad-band bus on the Space Station would permit bus media sharing while separating the DMS and PMAD data streams. The desire for a common bus stems from hardware commonality and elimination of duplicate cables for DMS and PMAD and also to permit the DMS and PMAD buses to coexist without interfering with each other. The need for separate buses comes from the fact that CSMA/CD networks such as ETHERNET and IEEE 802.3 also can experience a momentarily slow response time (access time) particularly when several other users elect to transmit long messages, one after the other, thereby denying access to a user that is in a hurry as in a real-time control system such as PMAD. This is worsened by the added time for collision resolution. Even deterministic buses such as IEEE 802.4 would have to set some of their own protocol to reduce permissable message length maximums for guaranteed rapid response. A broad-band bus would potentially and actually permit the DMS network and the PMAD control network to coexist as separate networks on the same cable at different frequencies, thereby eliminating message interference between networks. Adoption of a broad-band bus would permit parallel development of DMS and PMAD with little or no interaction occuring between the two systems. There is no question that such an approach would do the job; however, the remaining question would be one of cost and practicality (weight, size, maturity, etc.)

There are commercial systems which perform in the above described manner which are in existence and could be adapted to a space environment. Each node which interfaces with the broad-band bus would have to contain a broad-band modem to transform the base-band network signal to a relatively high frequency carrier, typically 250 mHz, at a bandwidth of about 10 mHz. These modems cost approximately $3500 apiece and are about 1.3 cu ft in size and weigh about 15 pounds in existing commercial equipment. A wide band modem for space application probably could be made for about $20K-$100k; occupying about .25 cu ft and weighing about 3 pounds and with a non-recurring cost of less than a few million dollars. It is fairly obvious that elimination of an extra few hundred feet of cable would be more than offset by the added cost, volume and weight of from 50-100 modems, one at each node of the DMS and PMAD systems on the common bus. The broad-band bus would in no way improve or hinder the operation or response of the DMS or the PMAD Control buses as individual buses. If the concern about separation of major users of a common media bus is strong enough to bear the cost, then a broad-band bus is a workable solution.

7.13.12 Use of Fiber-optics

The practical use of lightweight fiber-optics cable as the media is still questionable ,particularly in a space environment. The ability to "tee" fiber-optics is both lossy and unreliable at the present state-of-the-art. Fiber-optics is basically a one-way media with two paths required to comprise a duplex circuit. Fiberoptics "stars" are used as the only practical way to accomplish a "tee" function but increase average number of cable runs and complexity as well as to increase signal amplitude dispersion.

Collisions are more likely in a fiber-optic system due to differences in cable length and in transceiver outputs particularly when multi-point star couplers (which are inherently lossy and variable between taps) are used. Received optical signals can vary in amplitude by as much as 10 dB or more. With that much deviation, a high amplitude signal could mask a low amplitude optical signal so completely that an impending collision would be impossible to detect. The use of repeaters, bridges and optical-to-electronic converters further adds to signal amplitude variation.

Fiber-optics connectors also still are quite unreliable, large and relatively heavy which further mitigates against the use of fiber-optics at this time except for long-haul point-to-point cables in excess of a kilometer in length or in special EMI/radiation situations or in applications in which extremely wide band widths are required. In ground environments fiber-optics connectors get dirty causing signal loss or even stoppage. The long-term effects of micro-dust particles in space or contaminating fiber-optics junctions should be studied before adopting fiber-optics. Also, the effects of extreme cold temperature on light refraction/transmissivity within the fiber should be studied. Stability of adhesives which are used as joining mediums with many connectors should be investigated. It is possible that the state-of-the-art in fiber-optics will advance enough in the next few years to make this choice less risky. Welded joints rather than connectors would eliminate the dirt and reliability problem, however, how would you weld the add-on for growth or repair damage in space?

7.13.13 Recommended Base-line

1) The use of local area nets for the RPC/RBI interface using parallel I/O or IEEE 488 as the RPC/RBI interface in PMAD if parasitic power is not too costly; the use of MIL-STD-1553B if extremely rapid response-time is not a predominent requirement and parasitic power is very costly. Since parasitic power is costly, we recommend and will baseline 1553 for local area networks.

2) The use of either something similar to IEEE 802.4 or MIL-STD-1553B as the processor-to-processor interface in the PMAD Control net. Since 802.4 components are not readily available, we recommend and will baseline 1553.

3) The use of a common broad-band coaxial cable network allowing PMAD and DMS to share cables if concern about interaction is strong enough to warrant the added cost. (A NASA decision).

The analysis must be continued to include redundant paths and the ability to survive damage by the incorporation of fault-tolerant architectures.

7.14 BATTERY AND ARRAY SIZE SELECTION

The key trade study performed in support of the PV subsystem definition was concerned with the sizing of the array power and battery capacity. The major consideration was the optimization of array and battery for the polar platform while meeting the station requirements in a cost-effective manner. Emphasis has been placed on Polar Platform optimization to minimize its first launch mass.

In addition to the general requirements stipulated in the Space Station Program Power System Definitions and Requirements, specific ground rules used for the battery and PV array size trade are:

- Minimize polar platform first-launch and IOC EPS mass.
- Identical assemblies on platform and station for source hardware (strict commonality).
- The platform carries one redundant battery at first launch and IOC.
- Platform battery DOD is 35% maximum with one battery out.
- The station must have even number of battery modules but carries no redundant batteries.
- Station battery DOD is 35% maximum with all batteries working.

In addition to these criteria, the following are considered as desirable goals:

- The IOC platform has at least three batteries plus one redundant battery.
- Station PV nominal power capability of 25 kW at user input plus 1 kW for PMAD processors at 3 years in orbit.
- Minimize PV subsystem mass on the station.

Approach

Following detailed definition and refinement of array degradation factors for the worst-case altitude conditions for the polar platform and the station, the power capability of array panels for the 10-year polar orbit case and 3-year station orbit case were determined. A simple linear mass model equation was used for one array wing:

Wing mass = 181.2 kg + 5.5 kg/(panels/blanket)

and for one battery:

Battery mass = 72 kg + 2.4 kg/Ah.

These models have good validity in the ranges of interest.

The peaking requirements on the platform permit variation of depth-of-discharge carry-over to subsequent cycles, so long as full recharge is achieved at the completion of the two peaking orbits and two make-up orbits. Larger arrays minimize this carry-over, thus reducing the battery size required to maintain a maximum 35% DOD. Smaller arrays necessitate increased battery size.

Total battery capacity requirement for the station is a function of array capability. Since the platform-optimized array man not meet the station PV power goal of 25+1 kW, the batteries are not necessarily sized to support 25+1 kW, but rather the actual capability of the PV system up to 25+1 kW.

Trade Summary

Capacity Range - Potentially viable battery capacity options cover the range of 30 to 120 Ah. However, the platform redundancy considerations result in significant penalties at the very large capacity sizes. Thus, capacities above 80 Ah were effectively eliminated from consideration. Below about 50 Ah capacity, the quantity of batteries on the station becomes quite large and this begins to present significant cost penalties. Since appropriate redundancy levels for the platform do not demand batteries smaller than 50 Ah, the lower limit was set at that level for further trades. The viable range of 50 to 80 Ah was selected.

Array/Battery Trade - Figure 7.14-1 show the mass trade for 46 and 48 panel arrays and the selected parametric range of battery capacities. The mass figures represent the array wings, batteries, and the power-independent mass of the charge and discharge power converters (the power dependent part does not vary with capacity selection). The sawtooth shape of the curves reflects the modularity of the batteries: mass increases with capacity as batteries are increasingly oversized with respect to the need, until the point is reached where a smaller whole number of batteries fits the requirements (a whole even number in case of the station).

Array Size Selection - Only in the case of the IOC platform would a system with a 48-panel array be potentially lighter than a 46-panel system over a small range of battery capacities. In that range, there would be overall mass penalties on the first-launch platform and the station. With the first-launch platform being particularly mass-critical, the 46-panel array was selected as common baseline for the platform and the station, even though the station PV power capability does not quite reach the output power goal of 25+1 kW.

Battery Capacity Selection - Figure 7.14-1 illustrates the following potential capacity selections (46-panel case):

- 55 Ah - minimal first-launch platform mass
- 58 Ah - low station mass
- 62 Ah - minimal IOC platform mass
- 77 Ah - minimal station mass, minimal total installed mass

The 77-Ah option appears attractive, but the first-launch platform mass penalty is large. The 55-Ah capacity is best for first-launch but yields large penalties on IOC platform and station. The 62-Ah capacity, which provides low total installed mass with minimal added mass on the first-launch platform, is an attractive compromise. The 62-Ah batteries are therefore selected as baseline.

Battery Cell Diameter - Options are the conventional 3.5-inch diameter NiH_2 cells versus the more recently demonstrated 4.5-inch versions. The 4.5-inch cell, while nearly equivalent in maturity to the smaller size, is not mass-effective below about 90-100 Ah capacity. Since platform redundancy requirements force the selection of the cell capacity well below this level, the 3.5-inch diameter technology is a logical choice.

Battery Voltage and Modularity - Options are voltage levels optimally matched to the source bus voltage, and voltage levels that provide the opportunity for commonality with lower-voltage space station elements, such as the Mobile Servicing Center (MSC) and related systems such as the OMV. The former would require approximately 100 cells in series, split in two or more manageable series assemblies. The latter could be implemented with a modular battery assembly with 22 to 23 cells. Four of these in series (88 to 92 cells) would serve station and platform, and a single assembly would be compatible with low-voltage (MIL-STD-1539) systems.

The modular approach does not appear to present significant cost penalties, since practical constraints already dictate a level of physical modularity for the 100-cell batteries. Using a 23-cell modular design approach, battery development costs would be virtually eliminated for separate low-voltage systems, so that the overall cost of energy storage hardware for the program could be reduced. Therefore, a 23-cell battery assembly has been baselined as the building block for the space station and associated batteries.

7.15 PMAD ORUs PACKAGING

7.15.1 Methodology

The flow chart shown in Figure 7.15-1 reflects the approach used in performing this trade-off study. Once the design requirements were developed, the various possibilities of sizes were evaluated, and the conclusion served in the process of selecting the attachment to the utility plates. The selection of ORU attachment method affects both structural attachment and compatibility with EVA and telerobotic servicing. The source of method of heat removal micrometeroid protection, cosmic radiation and EM field protection, and orientation studies were run in parallel in order to establish the information base for the final selection. The study is preliminary, however the results obtained indicated the sound feasibility of the concept selected.

7.15.1.1 Objective

Obtaining preliminary baseline information on the major factors affecting PMAD packaging configuration and structure.

7.15.1.2 Assumptions

- o The functional components and/or quantities within an ORU are a given rather than a variable for these studies. Further optimization studies will be conducted during Phase C/D.
- o Penetration of the wall of an ORU by a micrometeoroid or orbital debris particle results in ORU failure.
- o WP-04 PMAD ORUs inboard of the alpha joint will govern the physical elements of the external interface with WP-02 utility plates.

7.15.2 Requirements

7.15.2.1 Electrical/Electronics Content

- o The content of the ORU should be provided with proper heat removal in order to maintain their temperature at an acceptable range.
- o Protection from the natural environment should be provided, specifically protection from micrometeoroids, space debris, cosmic radiation and EM field.

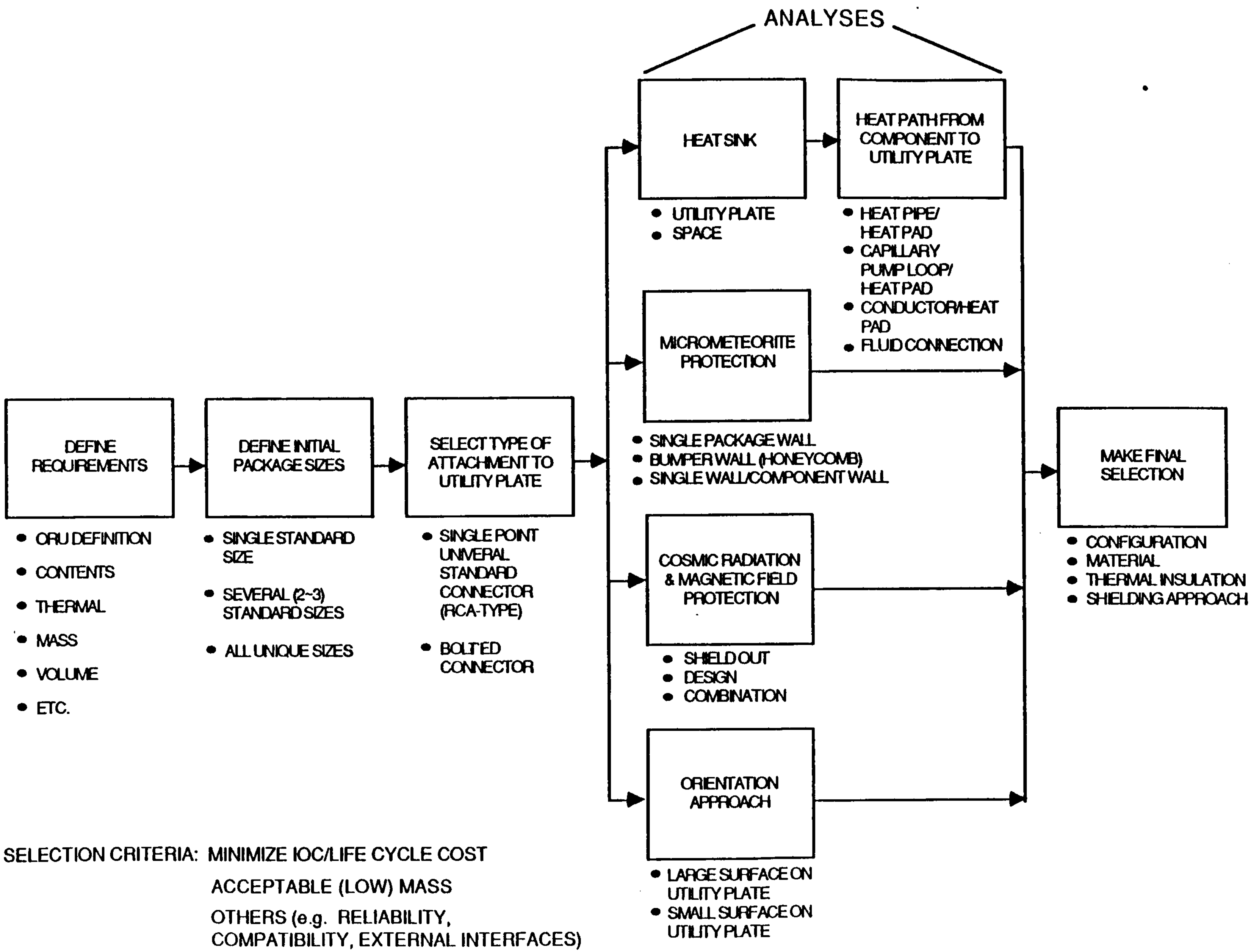


FIGURE 7.15-1 – PMAD-ORU PACKAGING : TRADE OFF STUDY–FLOW CHART

- Ease of access to the components during installation and maintenance.

Note: The issue of EMI/EMC was not part of the study.

7.15.2.2 Interfaces

Each ORU requires connections for electrical power, DC control power, data, heat transfer and structural attachment. The interface method selected should have the following characteristics:

- Compatible with EVA removal/replacement
- Compatible with the MSC and effector and with telerobotic removal/replacement
- Fulfill commonality requirements
- Constitute all the interfaces with 100% reliability for the required number of mating/demating.

The ORU envelope selected, in addition to providing protection against the natural environment, must further satisfy requirements pertaining to:

- Structural characteristics
- Ease of accessibility to internal components
- Ability to withstand ground handling and storage

7.15.3 Generic Analyses

The generic analyses (see Figure 7.15-1) were performed in parallel to ensure that each area considered the widest range of options applicable to the field, reserving resolution of any conflicts to a later phase of the study.

7.15.3.1 Sizes, Configurations and Orientation

Each ORU sizing was based on estimated components dimensions, mass, mounting geometries, and population. Two, actually, conflicting requirements were used - maximizing commonality, and percent occupancy. The study resulted in three standard case sizes for external ORUs and a common mounting method for internal ORUs. See Table 7.15-2. The interfacing element in all the ORUs is

Table 7.15-2

Standard Packages Size and Present Occupancy

ORU	Total Components Volume (in ^{3})	Package Size	% Occupancy
Transformer	248	A	8.4
Power Source Controller	414	A	13.9
AC Switching Unit	640	A	20.5
PV Controller	510	A	17.2
SD Controller	1020	B	25.2
MBSU	1156	B	28.5
DC Switching Unit	1548	B	38.2
PDCU (Truss)	1885	B	46.5
Battery Assembly	3960(*)	B	98
DC-AC Inverter	3150	C	38.3
PV Control Unit	2880	C	35.0
BCDU	3372	C	41.0
Frequency Converter	4200	C	51.1
NSTS Power Converter	7276	C	88.5
PMC	372	I	
NBSU	1266	I	
PDCU (Module)	1185	I	

$$\left.\begin{array}{l} A = 23 \times 25 \times 12 \text{ in} \\ B = 23 \times 38 \times 12 \text{ in} \\ C = 28 \times 38 \times 12 \end{array}\right\} \text{External Envelope Dimensions}$$

I = Internal rack mount

* Envelope of the 23 cells cluster

of the same size although the number of electrical power, DC and data connection may differ from one ORU interfacing unit to the other. Also after review of all the components dimensions, it was concluded that the height of all the ORUs can be the same. Another common feature to the PMAD ORUs is the bottom base plate to which the components are bolted. Component layouts for the highest density ORU in each class have been checked by Manufacturing to assure feasibility of production and maintenance.

The battery assemblies packaging is the same as for the PMAD packaging, and is therefore included in this study.

The issue of locating the components on the base plate only, or in combination with location on the walls or even the top was considered. Layouts were prepared for the individual ORUs as well as for the PV equipment box, utilizing both approaches. It was concluded that room availability, and access for maintainability made practically no difference regarding the location of the components inside the box. However, when considering heat pipe routing in the box, room for electrical and data harnesses, ease of access to the components and the overall weight of the ORU, the conclusion was that the base plate should be the only location for the components attachment.

Note: These conclusions were not reflected in the submitted layout drawings. Further studies will refine these conclusions.

7.15.3.2 Thermal

Preliminary thermal analysis was performed evaluating the following:

a) ORU passive cooling
b) Thermal interfacing with conductive pad (active cooling)
c) Thermal interfacing with fluid circulation (active cooling)

The requirement imposed, based on the battery assembly for cases b and c above, were 5 $\pm$ 5°C at the utility plate.

After eliminating 7.15.3.2-a due to various reasons (e.g. lack of full orientation control of the casing particularly during first two flights, weight and volume penalty, etc.) the concept of thermal interfacing with conductive pad between the ORU and the utility place was studied. The thermal resistance path from a component inside the ORU all the way to the radiator was investigated. See Figure 7.15-2. Thermal pads sizes and fit into the interfacing element were studied as well.

7.15.3.3 Natural Environment

High energy particle, EM radiation, and Macro/Micro collision environments were considered. The high energy particle environment consists of cosmic radiation and the solar wind flux which varies with solar activity. The polar orbit platform is also exposed to the flux of auroral electrons. The EM radiation environment consists of solar X-ray activity and manmade EMI.

The macro/micro collision environment consists of a sporadic micrometeoroid flux, with random arrival orientations and rate, a stream of micrometeoroid flux, with predictable arrival rates from well-defined directions, and an orbital debris flux, with a flux orientation along the orbital path and random arrival rate. Single wall and bumpered wall collision protection methods were examined, and a work list for further study of specific survivability considerations was developed.

7.15.4 Design Options and Selection

7.15.4.1 Sizes & Configurations

Three sizes have tentatively been selected, 23 x 25 x 12 in, 23 x 38 x 12 in, and 28 x 38 x 12 in, see Table 7.15-2. The actual design of the structural attachment shall be performed at Phase C/D. The baseplate is constructed of aluminum honeycomb to give the required rigidity without a large weight penalty and provide the proper heat sink for the components.

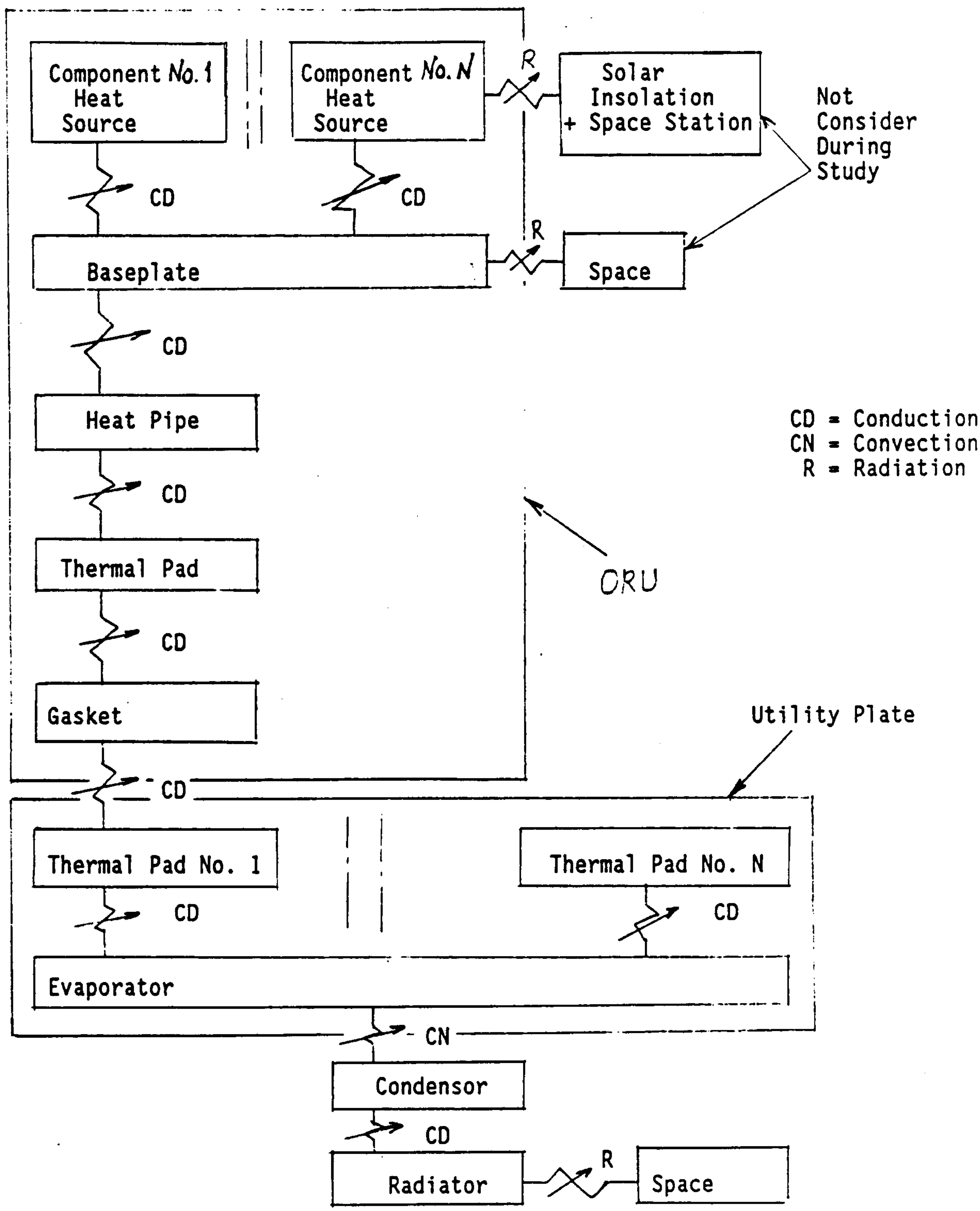


Figure 7.15-2 Heat Path: Component to Space

7.15.4.2 Thermal Dissipation

The Conductive pad type connector interfacing with the utility plate conductive pad is capable of conducting the necessary heat loads (less than 700W/ORU). A replaceable gasket is considered for use at the interface between the ORU and the utility plate conductive pads to overcome potential loss in conductivity due to minute unevenness in the interfacing surfaces, and avoid problems due to diffusion bonding between the ORU and utility plate. Convective cooling, utilizing fluid connections between the ORU and the utility plate were shown to be required for heat transfer rate above about 700 w/ORU. Hence, it was not used for the the ORU/Utility Plate interface.

Passive cooling of the ORUs was found unacceptable due to:

- o Lack of orientation control during initial installation
- o Increased envelope requirements/ORU
- o Increased mass/ORU

7.15.4.3 Shielding: Meteoroid & Debris

Collision protection studies indicate that a bumper-shield arrangement, in general, results in the least massive shield. Specific shield designs will be determined by the exposure-time product, geometry, and desired reliability of each ORU. For ORUs with higher reliability requirements a laminated main wall may be required. The outer walls of the ORUs must also protect the unit during ground handling and react torque loads during removal and replacement. These additional requirements may determine a lower bound on wall dimensions.

Bumper wall design is affected by the thickness of the inner wall, the outer wall, and the distance between them. See Figure 7.15-3. These variables are interrelated, and are dependent upon the colliding particle momentum. Optimization analysis was performed for bumper wall made of Al, for a survival probability over a 10 year period of .95. The results are shown in Figure 7.15-4. For comparison sake it should be stated that a single shield equivalent to the bumper wall would be twice as heavy.

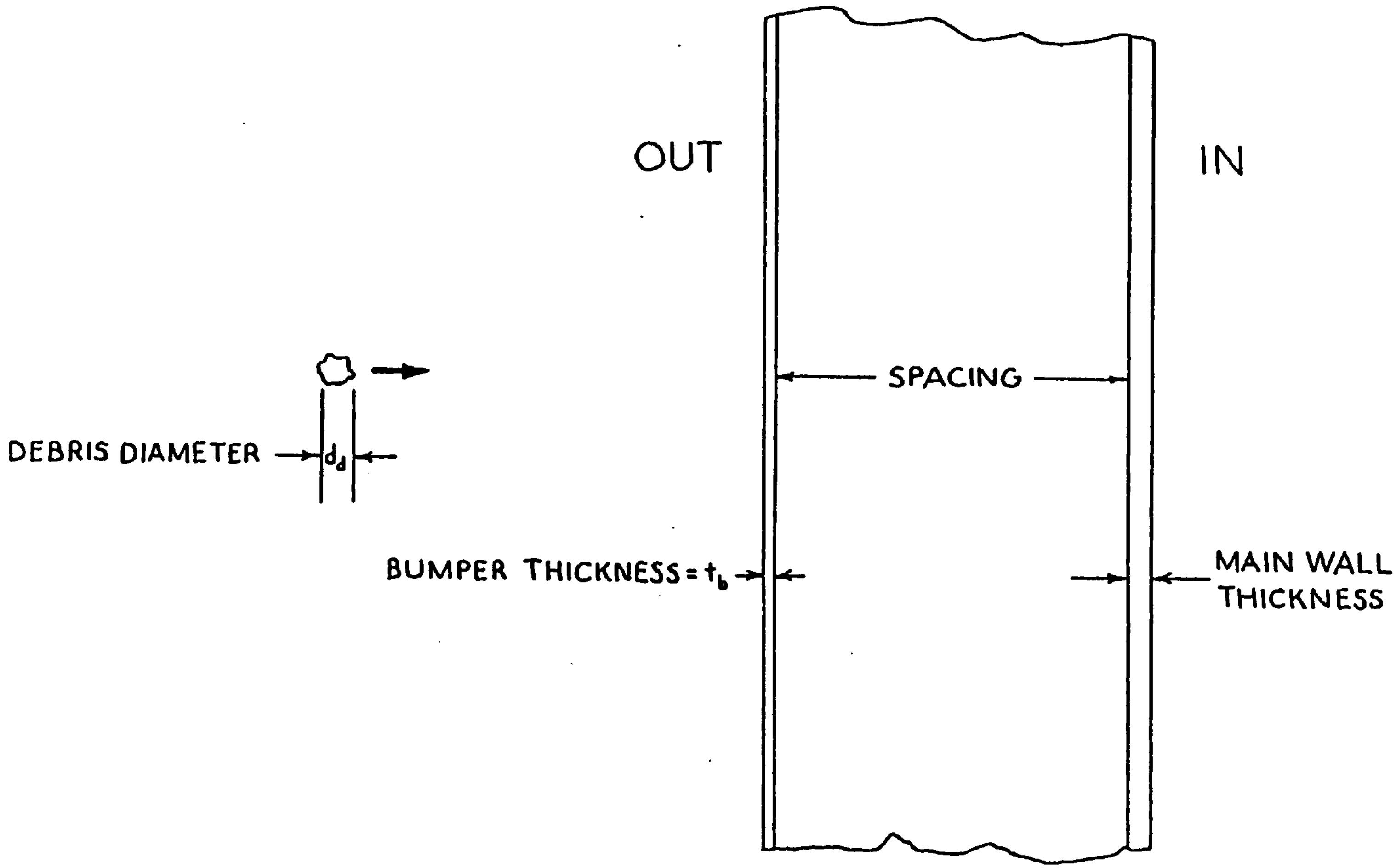
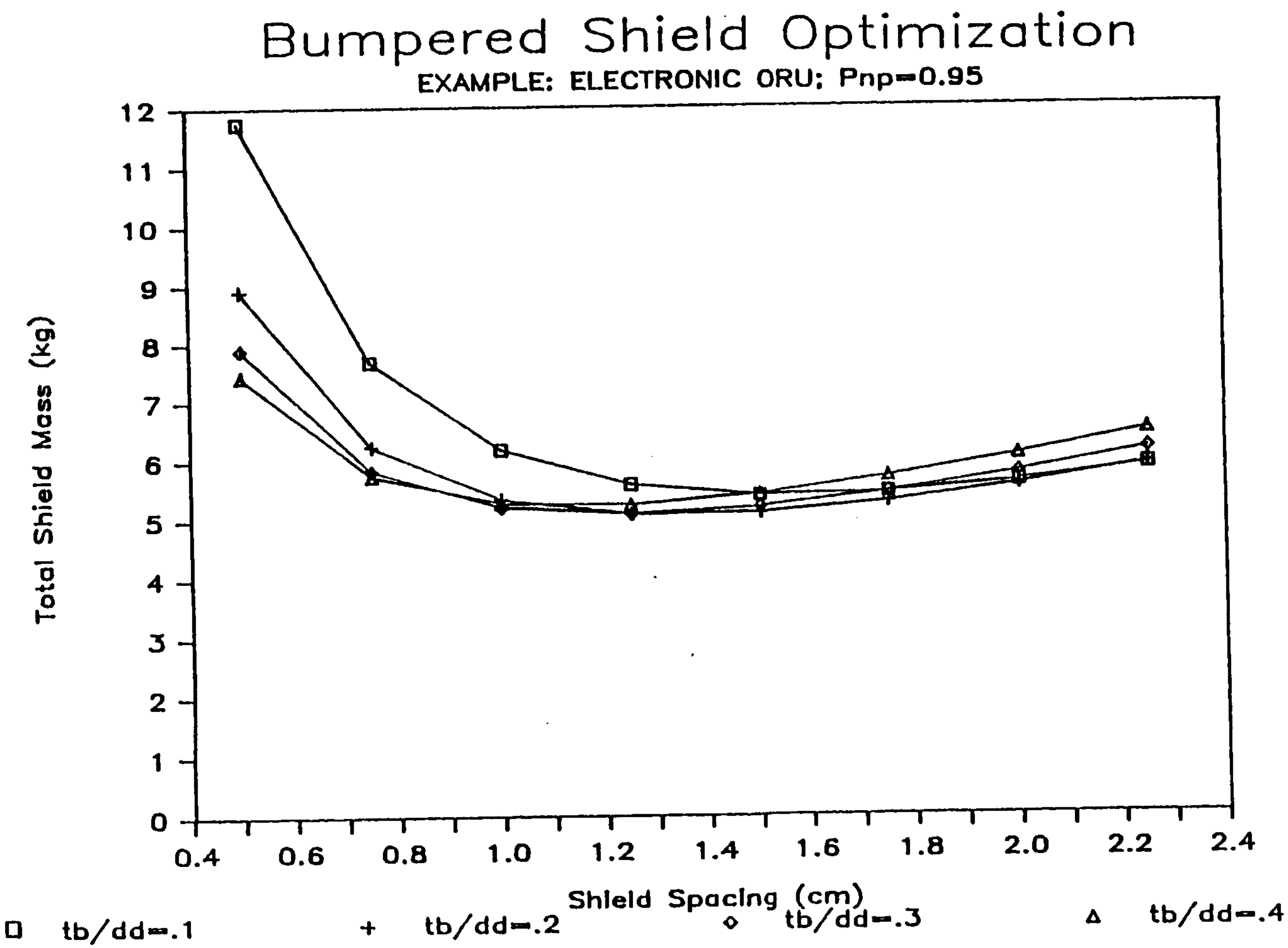


Figure 7.15-3. BUMPERED SHIELD GEOMETRY

FIGURE 7.15-4


Bumpered Shield Optimization
EXAMPLE: ELECTRONIC ORU; Pnp=0.95
Total Shield Mass (kg)
0
1
2
3
4
5
6
7
8
9
10
11
12
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
2.2
2.4
Shield Spacing (cm)
tb/dd=.1
tb/dd=.2
tb/dd=.3
tb/dd=.4

7.15.4.4 Shielding: Ionization

The conceptual evaluation of the subject of survivability resulted with the preliminary conclusion that inlieu of shielding all electronic components used in the Space Station and the platforms will have to be screened for radiation sensitivity, and some for tolerance to infrequent and randomly occurring upsets (due to heavy nucleids of the cosmic and galactic rays). Design techniques such as circuit circumvention, voting logic, etc. will need to be considered.

7.15.5 Conclusions

Figure 7.15-5 depicts the result of these conceptual/preliminary studies. The components are attached to a base plate, which has heat pipes embedded in it. The heat pipes are connected to conductive pads, which are mounted in the interfacing element. The interfacing element institutes the structural, thermal, electrical, data and control interfaces - using a single motion. The interfacing element also interfaces with the robotics end effector. The base plate and the interfacing element are enclosed in the bumper wall box walls and top, and are mounted on a utility plate.

FOLDOUT FRAME

FOLDOUT FRAME

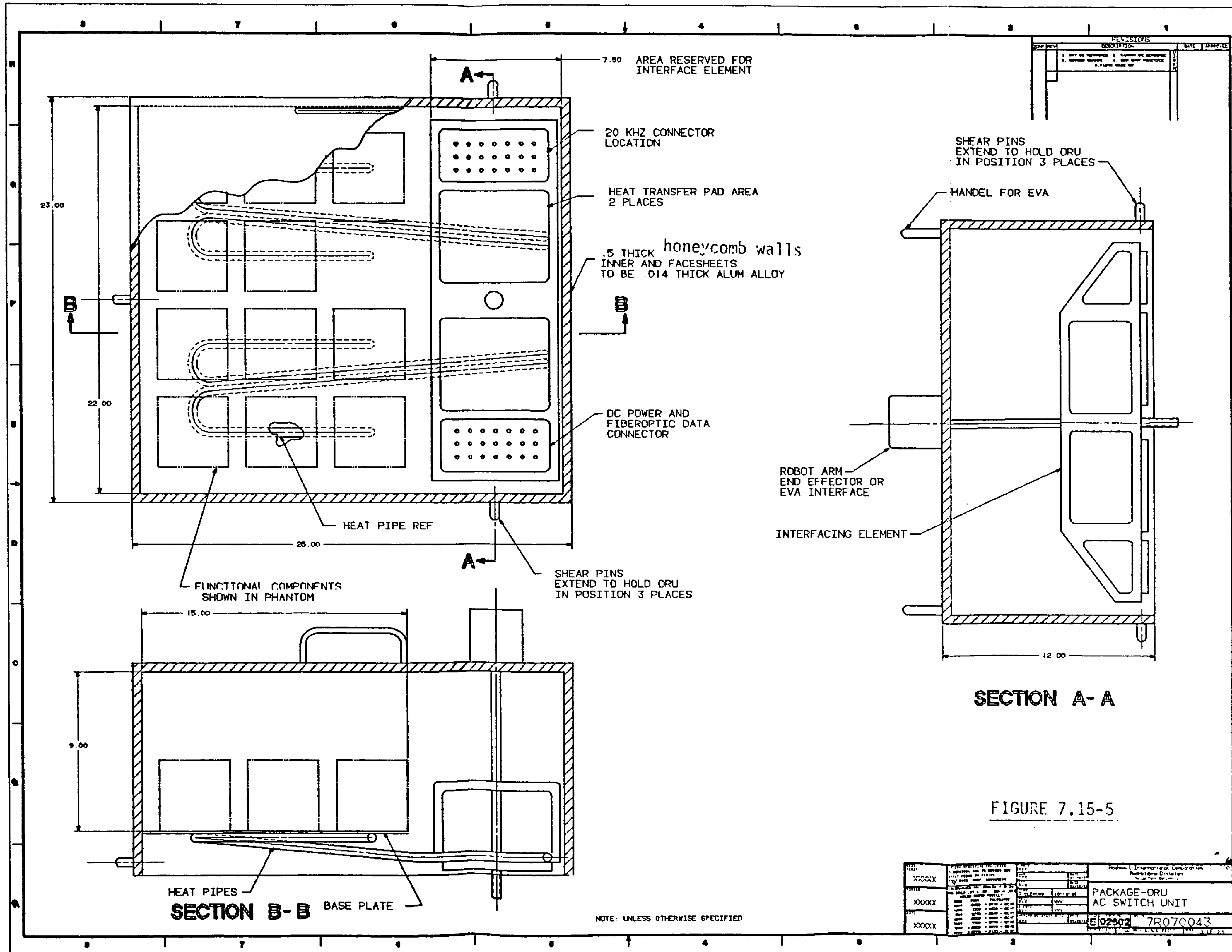


FIGURE 7.15-5

7.16 CAPILLARY PUMPED LOOP THERMAL CONTROL SYSTEM STUDY

7.16.1 Introduction

The baseline design of the integrated thermal control (ITC) system for the PV modules is described in section 2.1.1.5. The mechanically pumped two phase (MPTP) thermal transport system was selected. A study was performed in order to compare the MPTP system with the capillary pumped loop (CPL) concept. Both are two phase heat transport systems using ammonia as the working fluid. The primary difference is that the MPTP design incorporates a motor driven pump, while capillary action provides the pumping power in the CPL system. The conclusion was that, although the CPL system is better from a technical standpoint, commonality with the WP-02 thermal transport system favors the MPTP design.

7.16.2 Description Of CPL System

The ITC, shown schematically in Figure 7.16-1, is a redundant capillary pumped loop (CPL) system which uses ammonia as the working fluid. Alternate independent capillary pump evaporators are manifolded to separate, independent flow loops. Similar, alternately manifolded, independent flow paths exist in the condenser. Each loop can carry the entire cooling load so that loss of fluid in a single loop will not effect battery or PMAD capability.

The capillary pumped loop design is based on the CPL technology developed by the OAO Corporation, Greenbelt, Maryland. The OAO cold plate (Figure 7.16-2) provides heat acquisition from the battery or PMAD electronics. Each cold plate consists of aluminum honeycomb containing the redundant axially grooved aluminum evaporators. A porous wick provides the required capillary pumping mechanism. The batteries and PMAD electronics are packaged into electronic boxes as orbital replacement units (ORU's). Section 2.1.1.5.1 contains a discussion of design details of the ORU's. Each ORU chassis is itself an aluminum honeycomb structure, and it contains embedded heat pipes

PV MODULE INTEGRATED THERMAL CONTROL

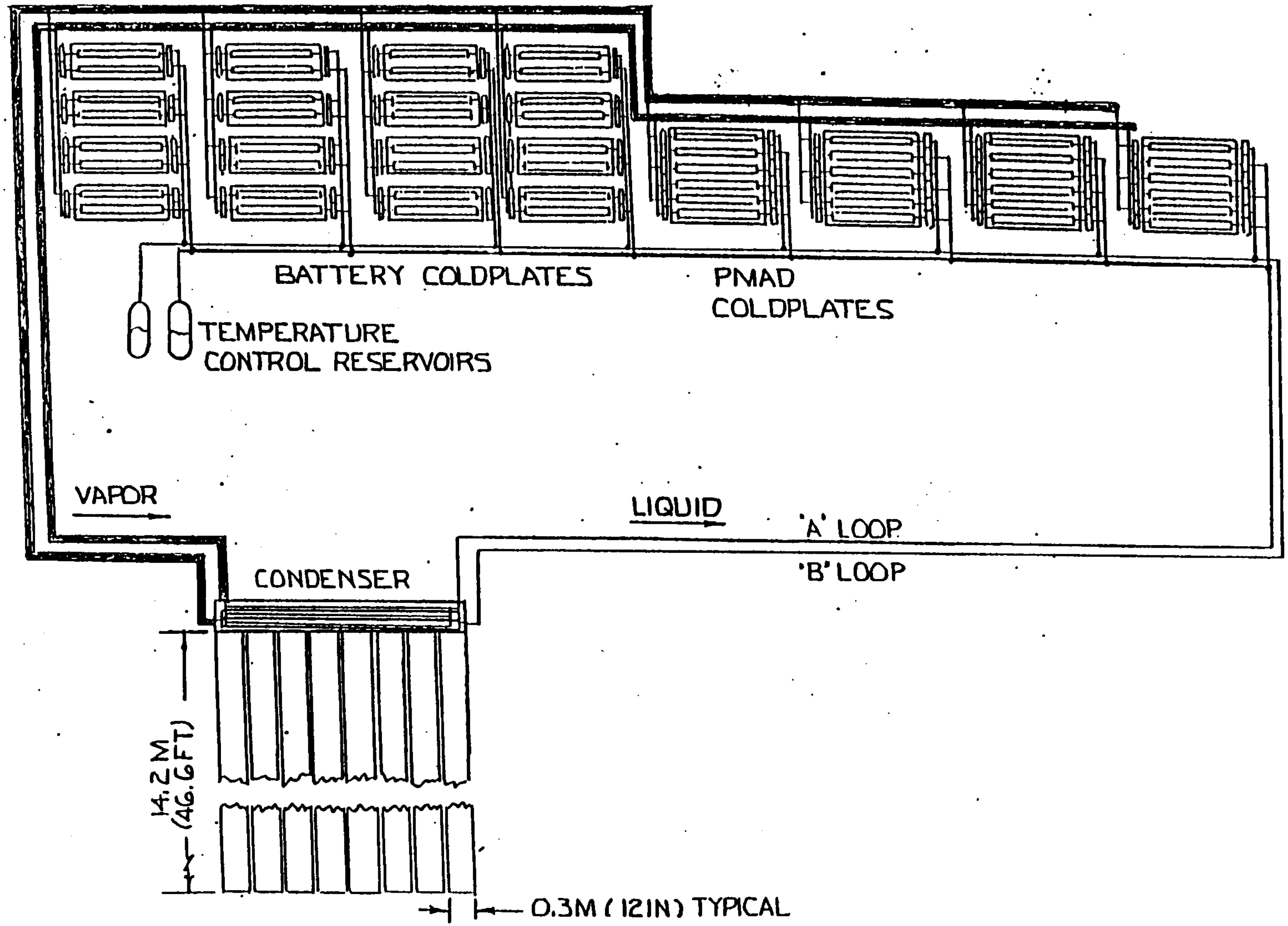


FIGURE 7.16-1

BONDED CPL COLD PLATE
(REDUNDANT)

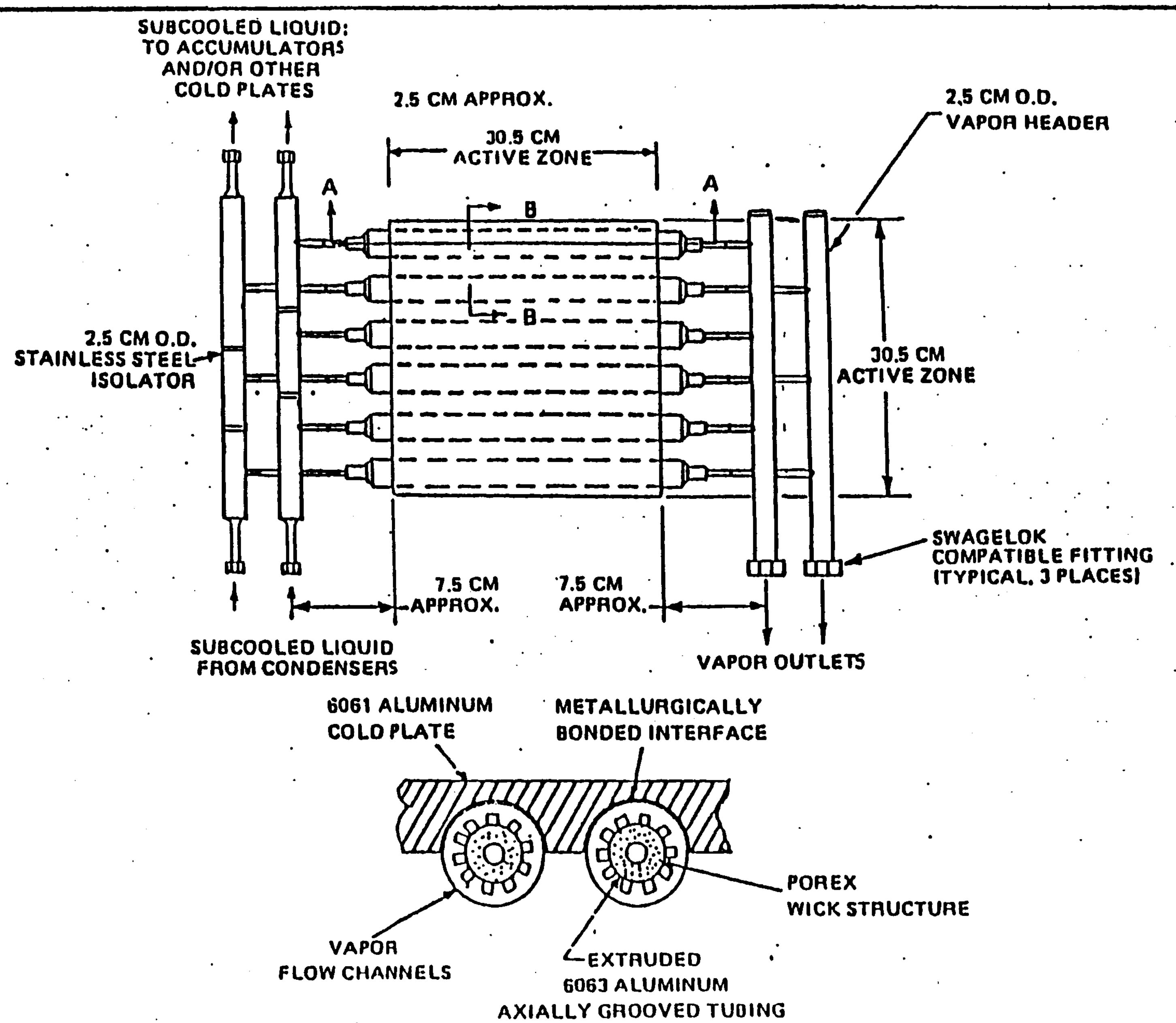


FIGURE 7.1.16-3

which transfer heat from the components to the chassis edge. The generated heat is then transferred from the heat pipe condensers to the CPL cold plate across a dry, bolted interface. The CPL cold plate is part of the system utility plate, which also contains interfaces for the transfer of data and power from the ORU's to other parts of the station.

7.16.3 Performance Definition

The thermal rejection system has been sized for the orbital average peak heat rejection requirement of the battery, despite the fact that there is considerable thermal mass in the batteries. The PMAD heat rejection is based on the maximum heat rejection of each ORU and the maximum number of ORU's that are operational at any one time. For a single module, the resulting total heat rejection required is 6.0 kW. The system is designed to reject this amount of heat with the CPL cold plates at 2°C. Selection of this cold plate temperature assures that the nominal 5 ± 5 C temperature is maintained at the batteries under all but contingency or failure conditions. The PMAD and battery cold plates are all maintained at the same temperature, even though the PMAD equipment can operate satisfactorily on cold plates maintained at a temperature of 20°C.

A two phase capillary pumped heat transport loop using ammonia as the fluid is used to collect and transport the heat from the PMAD and battery cold plates to the radiator system. A schematic of the capillary pumped loop design is given in Figure 7.16-3. This design has the advantage of requiring no moving parts and little power. The heat load on the cold plate evaporates the ammonia in a porous wick structure as shown in the capillary pump detail in Figures 7.16-4 and 7.16-5. The vapor is then condensed in the radiator heat exchanger. The capillary forces in the porous wick provide the pumping power to return the liquid to the capillary pump where it is again evaporated. A temperature controlled reservoir provides ammonia to flood the pumps for initial start up, insures that they are constantly receiving liquid at the inlet, and controls the temperature and pressure at which the loop operates. An isolator consisting of an annulus for liquid flow and a porous wick similar

CAPILLARY PUMPED LOOP SCHEMATIC

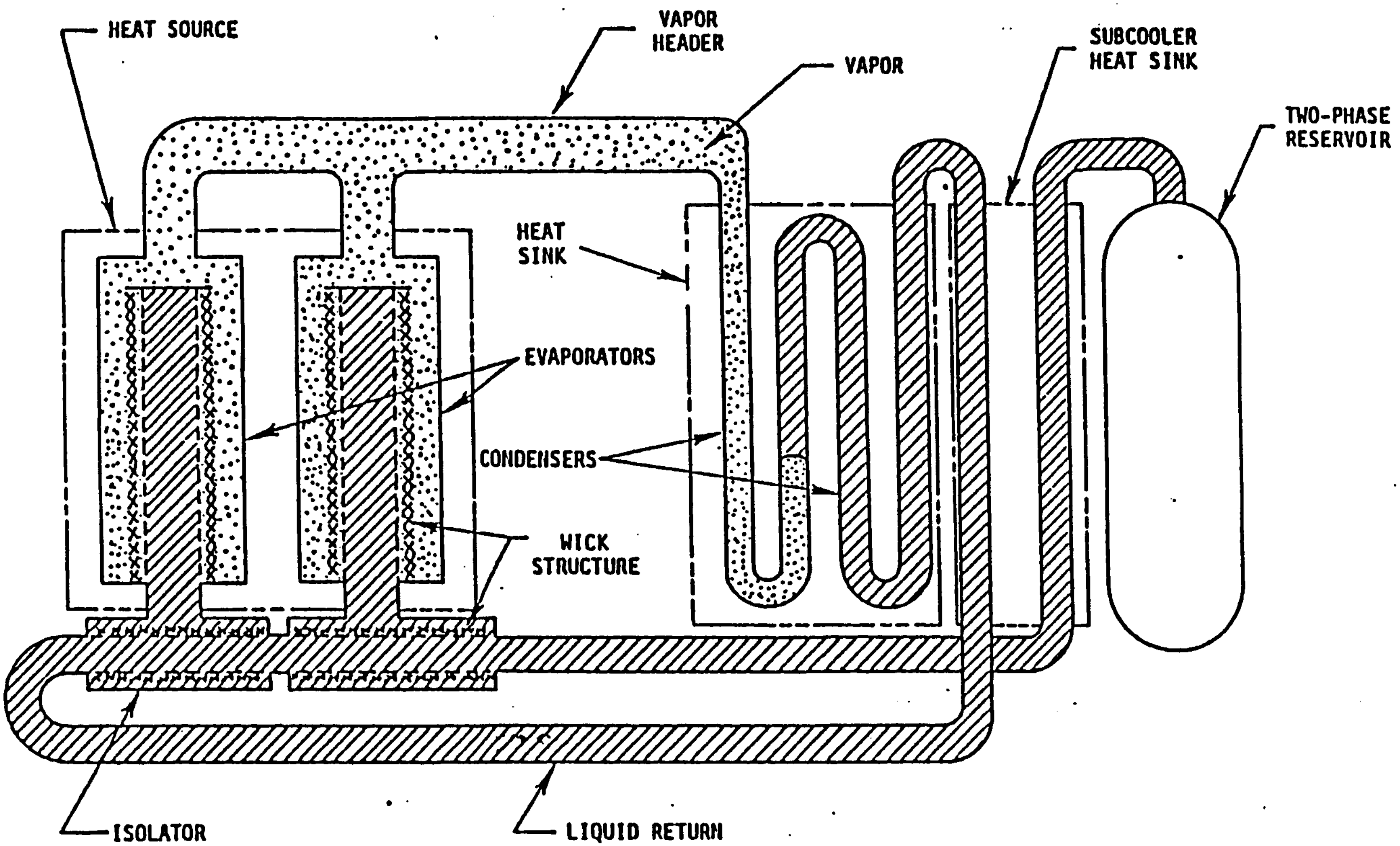


FIGURE 7.16-3

CPL EVAPORATOR HEAT AND FLUID TRANSFER

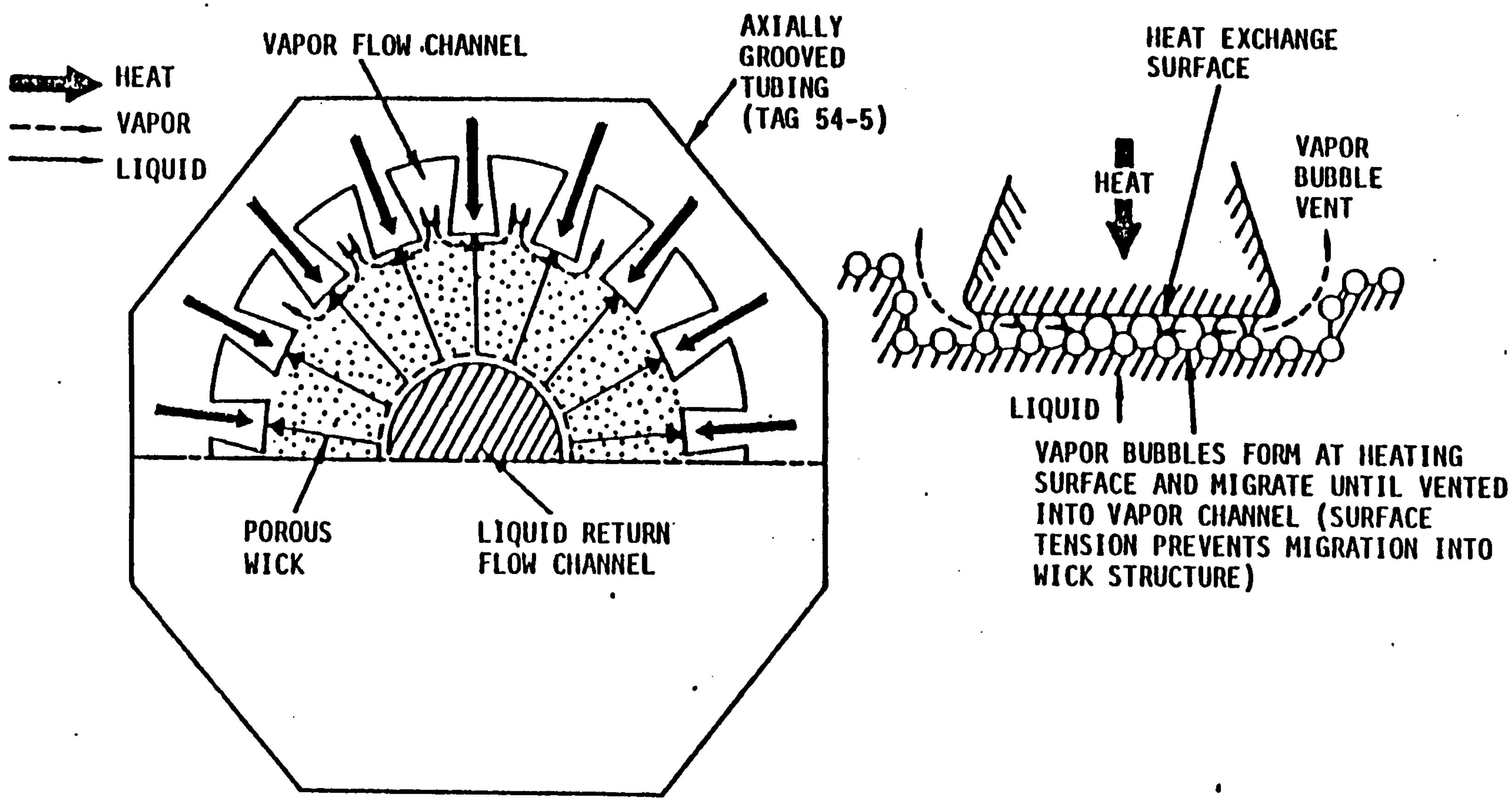


FIGURE 7.16-4

CAPILLARY EVAPORATOR PUMP

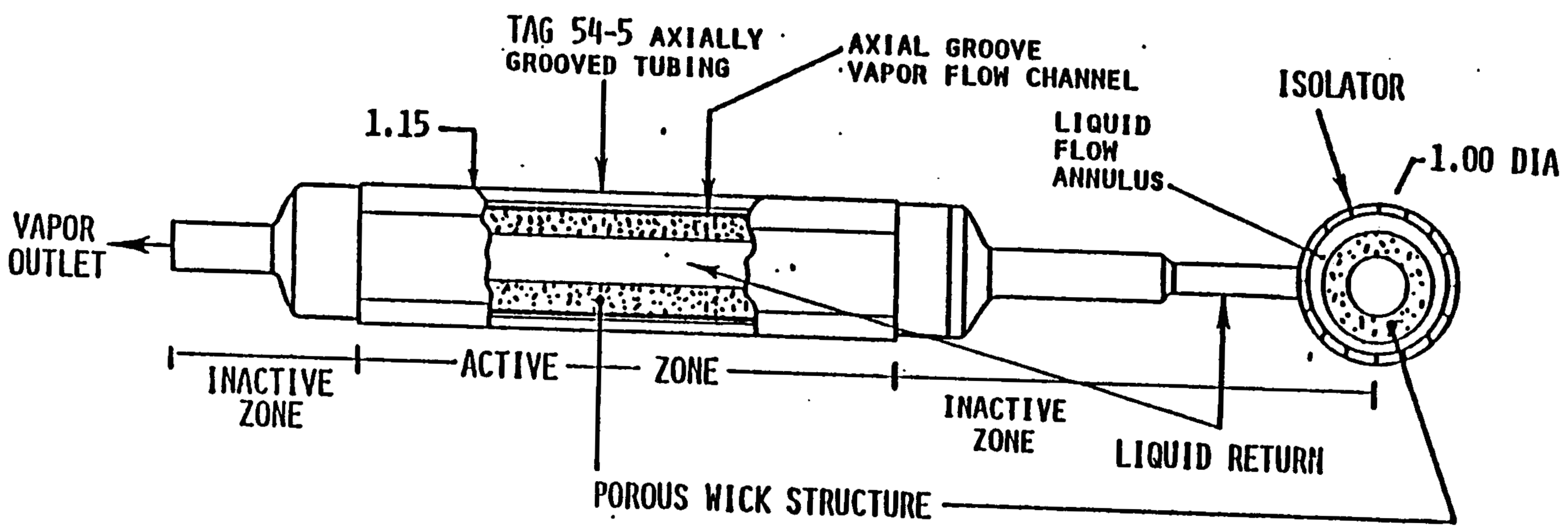


FIGURE [illegible]

to that in the pump prevents the depriming of one pump in the parallel flow arrangement from affecting the operation of the other pumps. In the event of depriming of an evaporator pump, or of a sudden change in the thermal load on one or more pumps, the reservoir control restores the CPL system to a stable operating mode. These operating characteristics have been demonstrated by OAO Corporation and NASA GSFC under both ADP programs and for WP-03 platform thermal control. Two engineering units, each capable of rejecting 6.3 kw of heat over a 10 meter transport distance, have been built and tested extensively. These tests demonstrated the transport limit, heat load sharing between evaporators, liquid inventory and temperature control by the reservoir, pressure priming under heat load, the ability of legs of the condenser to automatically shut down when they become too hot, and isolation of a single deprimed evaporator. Two smaller CPL systems have been flown on the shuttle on STS 51-G (6/85) and STS 61-C (1/86). Flight test results obtained during zero-g operation were almost identical to results of the same tests performed on the ground. From experience gained with these models, a vapor line diameter of 1.0 in and a liquid line diameter of 0.5 in were selected for the CPL system in each utility center. The maximum capillary pumping head developed will be approximately 0.5 psi.

7.16.4 Conclusion

The performance of both the MPTP and CPL systems has been demonstrated in testing at one g. In addition, two CPL systems have flown in the payload bay of the shuttle. These units verified that the performance of the CPL design in space is the same as on the ground. The MPTP design has yet to be flight tested. The CPL concept is inherently self controlling, and the absence of any moving parts makes it more reliable and less complex. The MPTP design has been selected as the WP-02 thermal transport system, due in part to the fact that the thermal transport distances inboard of the alpha joint are much longer than any that have been demonstrated using CPL. If the mandate to maximize commonality is overriding, the MPTP design will be favored.

7.17 ORC PARASITIC LOAD IMPLEMENTATION TRADE STUDY

A trade study was performed to determine the optimum methods of providing speed control for the PCU and power matching of the the PCU output with user loads. The trade resulted in a diode switched decimal related switched load being selected.

To maintain the PCU output power and frequency relatively constant, it is necessary to divert power not required by the customer to a parasitic load. This study compared the various approaches considered for the implementation of the control or switching of the load resistors. The design of the resistor/radiator was not considered.

Important requirements of the parasitic load are:

1. Sized to meet worst case maximum power output plus contingency margins.
2. Minimum "off" power consumption to maximize efficiency.
3. Minimum electromagnetic interference.
4. High reliability and low susceptibility.
5. Minimum harmonic distortion reflected to generator and minimum effect on user power quality.

7.17.1 Load Options

As shown in Figures 7.17.1a and 7.17.1b the turbine shaft can be loaded directly or the output generator can be loaded electrically.

Pro's and cons's of loading the turbine directly rather than through the customer output generator are:

- o Parasitic load controls will not affect customer waveform quality.
- o The field regulated generator output implementation has low component count and reduced stress on solid state components.
- o Inherent separation between speed control of customer generator must be fast enough to meet design goal.
- o Increased size and weight of turbine power diversion device must be considered.
- o Additional bearings needed - potential critical speed problems.

FIGURE 7.17.1a

POWER NOT REQUIRED BY CUSTOMER IS DIVERTED FROM THE TURBINE SHAFT

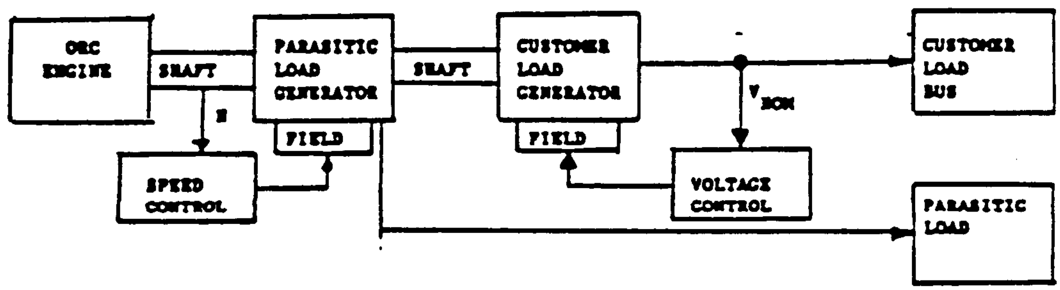
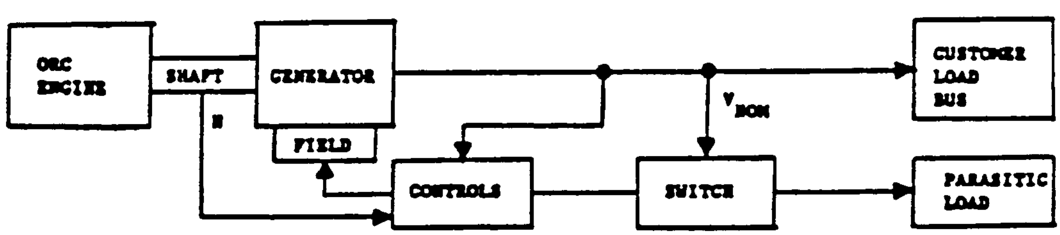


FIGURE 7.17.1b

POWER NOT REQUIRED BY CUSTOMER IS DIRECTLY DIVERTED FROM THE CUSTOMER SUPPLY LINES TO THE PARASITIC LOAD BANK



Direct loading approaches are shown in Figure 7.17.2 and are discussed in the following.

The turbine may be loaded directly by means of a dedicated parasitic load generator (controlled or uncontrolled output) or electromagnetic brake. The electromagnetic brake was rejected due to the problems of removing the heat. The dual permanent magnet generator (PMG) was rejected because of size, weight and mechanical risk. The electronic controlled load on the PMG is superior to the magnetic because of better response speed, size and efficiency but it still has many stressed components. Overall, a field controlled Homopolar or Rice Lundell generator feeding a fixed load was preferred because the field control system has low component stress and low component count. However, the direct loading scheme was rejected due to the added size and weight of the generator and the necessity for additional bearings. Accordingly the alternative approach of diverting power from the output of the generator was investigated.

Output generator loading approaches are discussed below,

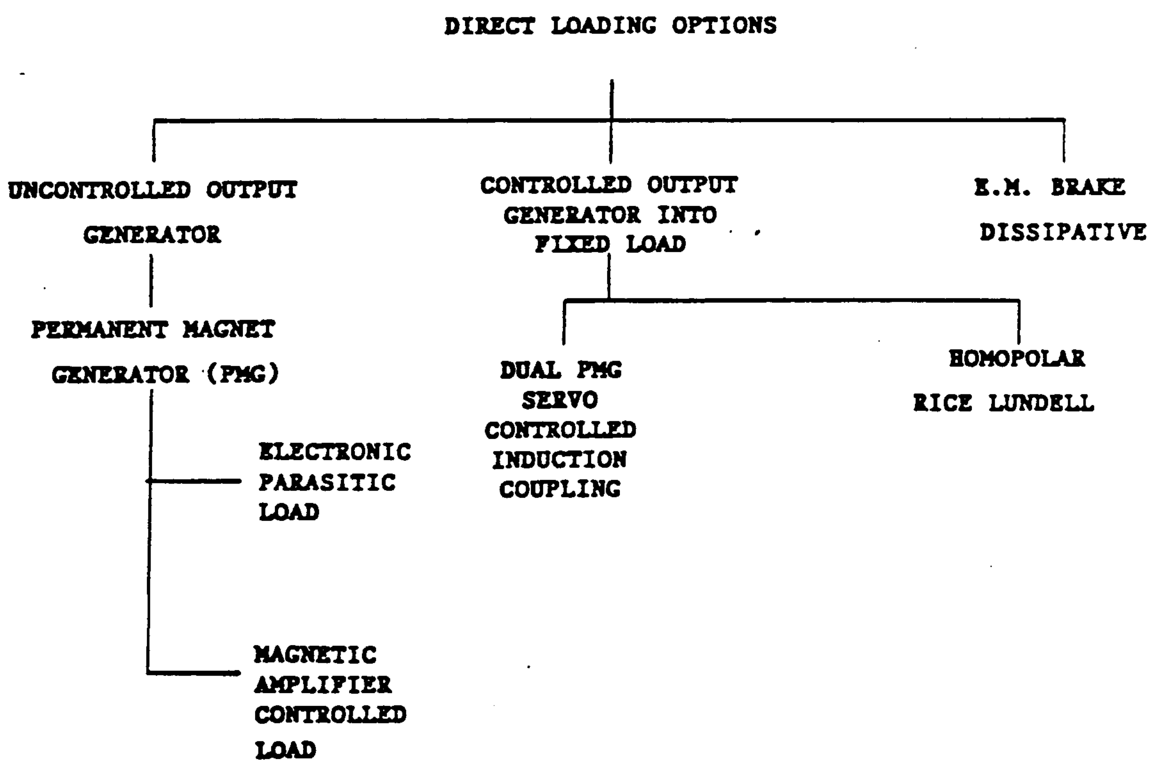
Motor generator set
(A motor which is electrically powered from the turbine output generator is mechanically linked to a second generator which is loaded by a fixed parasitic load. Load variation is achieved by varying the excitation of the parasitic load generator.)
- o Low distortion reflected to customer supply.
- o Unacceptable size and weight.
- o Slow response.

SCR Phase Control
- o Continuously variable control (non-linear transfer function near light load.)
- o Simpler and more robust than transistor option.
- o Higher EMI and current distortion away from full load.
- o Lagging power factor away from full load.
- o Frequency limitation.

Magnetic Amplifier Controlled Load
- o Similar to SCR phase control but more robust.
- o Slower than SCR phase control.
- o Increased size and weight over SCR phase control.

SCR Integral Control
- o Zero crossing firing: complete cycles.

FIGURE 7.17.2 Direct Loading Options



- o Less EMI but sub-harmonics generated.
- o Very slow response

Diode Bridge Linear Transistor Regulator
- o Minimal filtering: good response.
- o Drives different by probably as complicated as other approaches.
- o Needs heat sink capable of absorbing >15 kW but able to maintain junction below 105 C (MIL SPEC 1543)
- o Increased size and weight.

Diode Bridge PWM Transistor
- o Continuously variable control (linear transfer function).
- o Current distortion may be minimized by optimum switch patterns at expense of control complication.
- o Energy storage components and EMI filtering required which will tend to limit inherent fast response.
- o Complex drive scheme.

Diode Bridge Discrete Switched Loads
- o Control studies have indicated that a minimum load increment of 1 kW will give adequate speed resolution.
- o Lower EMI than PWM option since no high frequency continuous switching.
- o Simple on-off switches without the fast rise and fall times required for the PWM option yield simple drive circuits and lower losses.
- o Implementations possible which have minimum current distortion reflected to generator and output lines.
- o Minimal energy storage components allowing fast response to load transients.

Based on the previous discussion, the Diode Bridge Discrete Switched Load was selected.

7.17.2 Diode Bridge Binary vs Decimal Related Switched Load

7.17.2.1 Binary Related Switched Load

The switches used for the discrete switched load approach must be sized (including standard derating) for the maximum overspeed voltage. For a given resistive load the overspeed voltage also gives the maximum current and power losses which must also be within the derated switch ratings at the highest temperature.

This trade study was based on a 45 kW, 115 V. 0.75 P.F., 24000 RPM generator. It produces 79 V l-n into a 45 kW, 0.95 P.F. load and 184 VDC after 3 phase full wave bridge rectification. For a 30% overspeed the rectified DC voltage may reach 240 VDC.

A 45 kW total load at the normal speed voltage of 184 VDC may be implemented with five load resistors with binary related values plus one variable low power resistor as shown:

Resistance	1.5	3.0	6.0	12.0	24.0	24.0 to 480
Power (kW)	22.4	11.2	5.6	2.8	1.4	1.4 max
Current (A)	122.0	61.0	30.5	15.25	7.625	
Current (A) (@30% overspeed)	162.0	81.0	40.5	20.25	10.125	

Concerns with this approach are:

- Five different switch designs and load resistors and one variable load are required with attendant design, procurement and spares implications.

To avoid control problems the load must vary monotonically. To achieve this, the resistor ratios must track with equal temperature coefficients and drift. In absolute terms, 48 ohm (1/2 least significant bit) is equivalent to a 3% tolerance in the 1.5 ohm. these are common problems in D/A converters but, in this case, large powers are involved and variations in the ON resistance of the switches further complicates control.

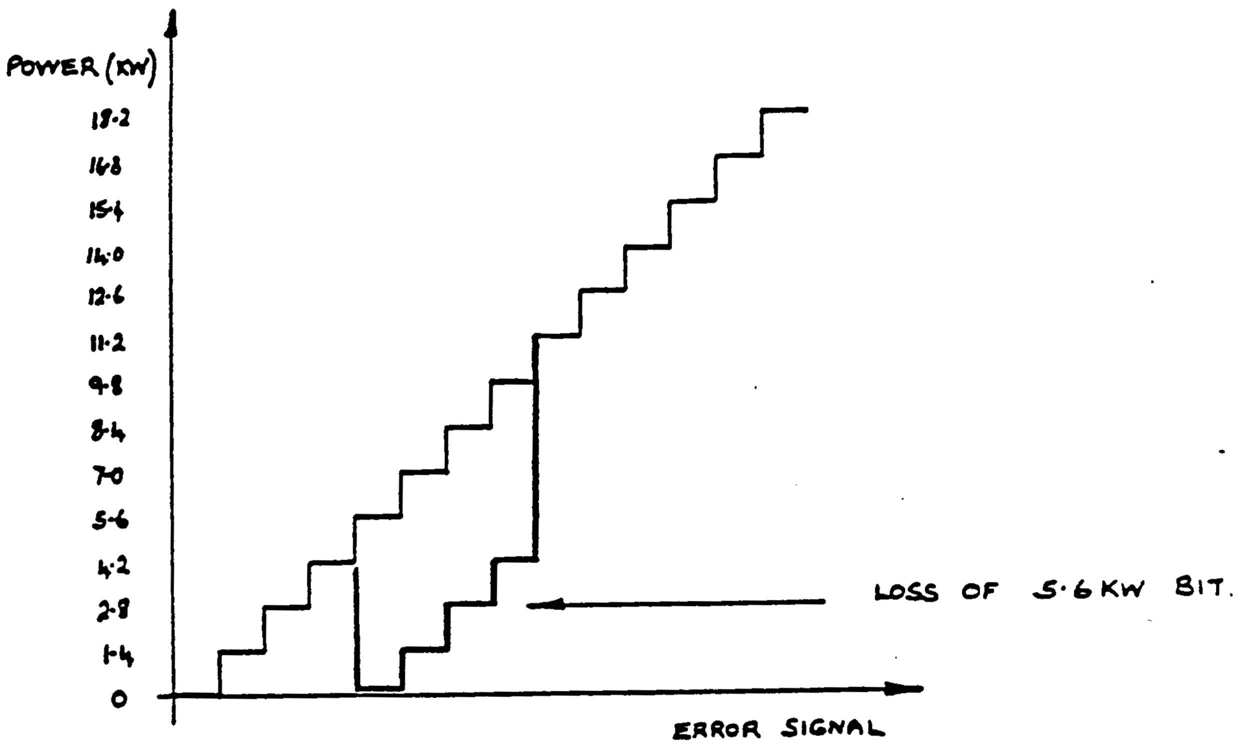
Table 7.17.1 demonstrates the binary switch operation over the full load range. It shows that the transitions between 9.8 kW and 11.2 kW, 21 kW, and 22.4 kW, 32.2 kW and 33.6 kW are very noisy as they involve switching several switches on and off for a small transition. In the case of the 21 kW transition the four least significant are switched off and the most significant switched on to increase the load to 22.4 kW.

Figure 7.17.3 illustrates, for example, the effects of losing the 5.6 bit, due to open circuit of the switch or load resistor which will cause control problems and result in insufficient total load to control speed.

TABLE 7.17.1

BINARY POWER SWITCHING

Switch Rating	2^5 22.4	2^4 11.2	2^3 5.6	2^2 2.8	2^1 1.4	Power (kW)
Noisy Transitions	0	0	0	0	0	0
	0	0	0	0	1	1.4
	0	0	0	1	0	2.8
	0	0	0	1	1	4.2
	0	0	1	0	0	5.6
	0	0	1	0	1	7.0
	0	0	1	1	0	8.4
9.8 - 11.2 kW	0	0	1	1	1	9.8
	0	1	0	0	0	11.2
	0	1	0	0	1	12.6
	0	1	0	1	0	14.0
	0	1	0	1	1	15.4
	0	1	1	0	0	16.8
	0	1	1	0	1	18.2
	0	1	1	1	0	19.6
21 - 22.4 kW	0	1	1	1	1	21.0
	1	0	0	0	0	22.4
	1	0	0	0	1	23.8
	1	0	0	1	0	25.2
	1	0	0	1	1	26.6
	1	0	1	0	0	28.0
	1	0	1	0	1	29.4
	1	0	1	1	0	30.8
32.2 - 33.6 kW	1	0	1	1	1	32.2
	1	1	0	0	0	33.6
	1	1	0	0	1	35.0
	1	1	0	1	0	36.4
	1	1	0	1	1	37.8
	1	1	1	0	0	39.2
	1	1	1	0	1	40.6
	1	1	1	1	0	42.0
	1	1	1	1	1	43.4



- LOAD MUST VARY MONOTONICALLY WITH SPEED ERROR TO AVOID CONTROL PROBLEMS

FIGURE 7.17.3

An approach to the first concern of five different switches and resistor load designs would be to use modular switches. Figure 7.17.4 shows switches of nominally 8A and 48A rating which have been used by Sundstrand in previous applications. The required switch loads could be made up as shown in Table 7.17.2. The 8A MOSFET switch is robust, simpler, has low OFF drive losses and high reliability. In this application where there is an advantage in making a load of many parallel switches the modular approach can be taken even further and six 8A MOSFET switches used in place of each 48A switch as shown in Figure 7.17.3. Initial packaging studies show the overall volume (including drive circuits and supplies) will not increase and spreading the heat load over six T03 can size footprints is advantageous.

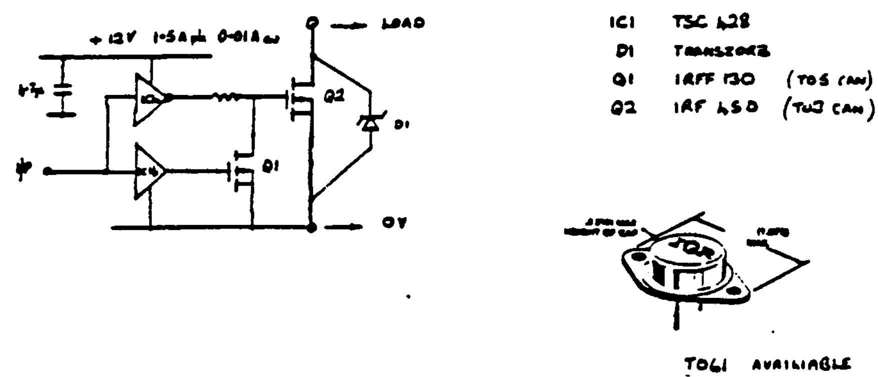
The load is then composed of multiple 8A switches and resistors grouped and controlled in binary increments. The malfunction of an individual 8A switch module, maintenance of binary resistor ratios and temperature coefficient ratio tracking are less of a problem but the noise and monotonic problems remain.

7.17.2.2 Decimal Related Switched Load

Rather than grouping the 8A switch and load modules into binary related groups, they can be individually switched in a decimal counting mode. The resolution is the same as the binary system and switching noise is limited to that of an individual 8A switched load. Resistor matching and tolerancing is not required and malfunction of any module will not cause monotonic problems (i.e. if a module is open circuit, the control will switch on the next module in the decimal count). Additional redundant units may be provided in this scheme.

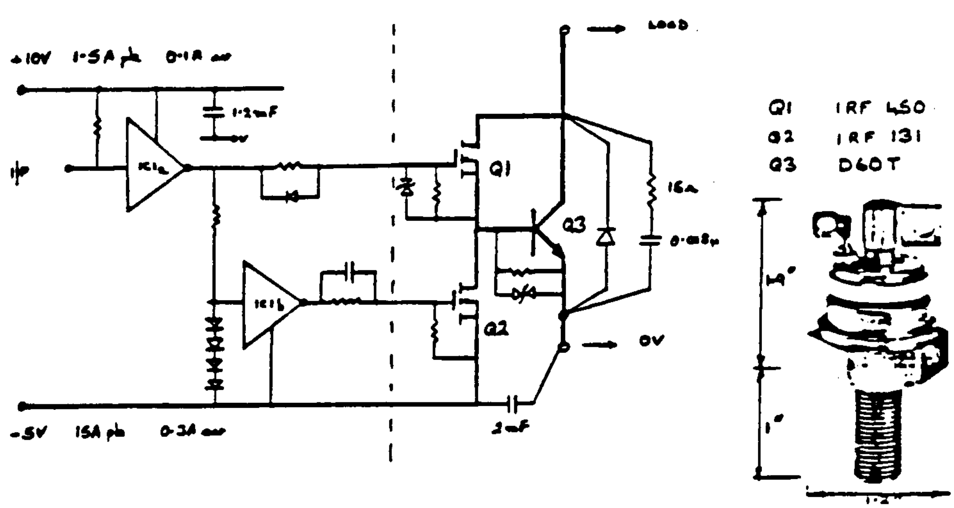
FIGURE 7.17.4

Typical 8A 450V (105°C Junction) MOS Switch and Driver



- o Simple drive circuit
- o Light snubbing
- o Very low power required to maintain device off.

Typical 48A 450V Switch and Driver Used on C141 Actuator



- o Complex drive circuit
- o Requirement for -5V supply to absorb stored base charge at switch off
- o Heavy snubbing
- o Q3 D60 requires Q1, Q2 to3 size devices for drive.

TABLE 7.17.2

BINARY LOADS MADE UP OF 6 OHM/48A AND 6 OHM/8A MODULES

BIT NO.	LOAD (ohm)	CURRENT (amp)	MODULE QUANTITY 6ohm/48 A	6 ohm/8A
1. (MSB)	1.5	162.	4.	-
2.	.3	81.	2.	-
3.	6.	40.5	1.	-
4.	12.	20.25	-	3.*
5.	24.	10.125	-	2.

TABLE 7.17.3

BINARY LOADS MADE UP OF 48OHM/8A MODULES

BIT NO.	LOAD (ohm)	CURRENT (amp)	MODULE QUANTITY
1.	1.5	162.	24.
2.	.3	81.	12.
3.	6.	40.5	6.
4.	12.	20.25	3.*
5.	24.	10.125	2.

* load changed to 48 ohm

7.17.2.3 <u>Conclusion</u>

The decimal approach with 8A switches was selected as meeting the requirements and preliminary design calculations were made.

At Unity Power Factor 30% Overspeed Bridge Output	270 V
Minimum Load Resistor 34 ohm at 8A	Power=2176 W
PLR Current at Normal Output of 184V/245A	Power= 45 kW
Current for 34 ohm switch at 184V/5.4A	Power= 994 W
Number of Switches Required: 46	

With 10% redundancy 50, 8A switches and 34 ohm loads are recommended.

Note that the worst case sizing approach results in less than half rated power operation of the switches and load resistor under normal operation. The proposed use of a microcomputer for control would also allow the randomizing of the switches to provide a more uniform operational life.

7.18 SD Radiator Sink Temperature Variations Study

Sink temperature plots were generated using a TRASYS model of the overall Space Station (see Figure 7.18-1) and a SINDA thermal model of the radiator and concentrator. The Space Station TRASYS model was run for three continuous orbits to generate radiator and concentrator environments and radiation form factors between the radiator and the concentrator.

These environmental fluxes and radiator conductors were incorporated into a SINDA model of the radiators and concentrator. A computerized model of the dual keel Space Station configuration is shown in Figure 7.18-2.

The concentrator thermal mass and front-to-back conduction was modeled based on the current preliminary design configuration. The radiators were modeled with an artificially low (near zero) thermal mass so that the resulting radiator temperatures represent variable sink temperatures.

Figure 7.18-3 shows the calculated radiator sink temperature variation during a single orbit for beta angle inclinations of -52, 0, and 52 degrees. The maximum sink temperature is seen to be about 212 K (-72°F) while the minimum value reaches 185 K (-126°F) with a beta angle of 0 degrees during the orbital eclipse period.

C-4

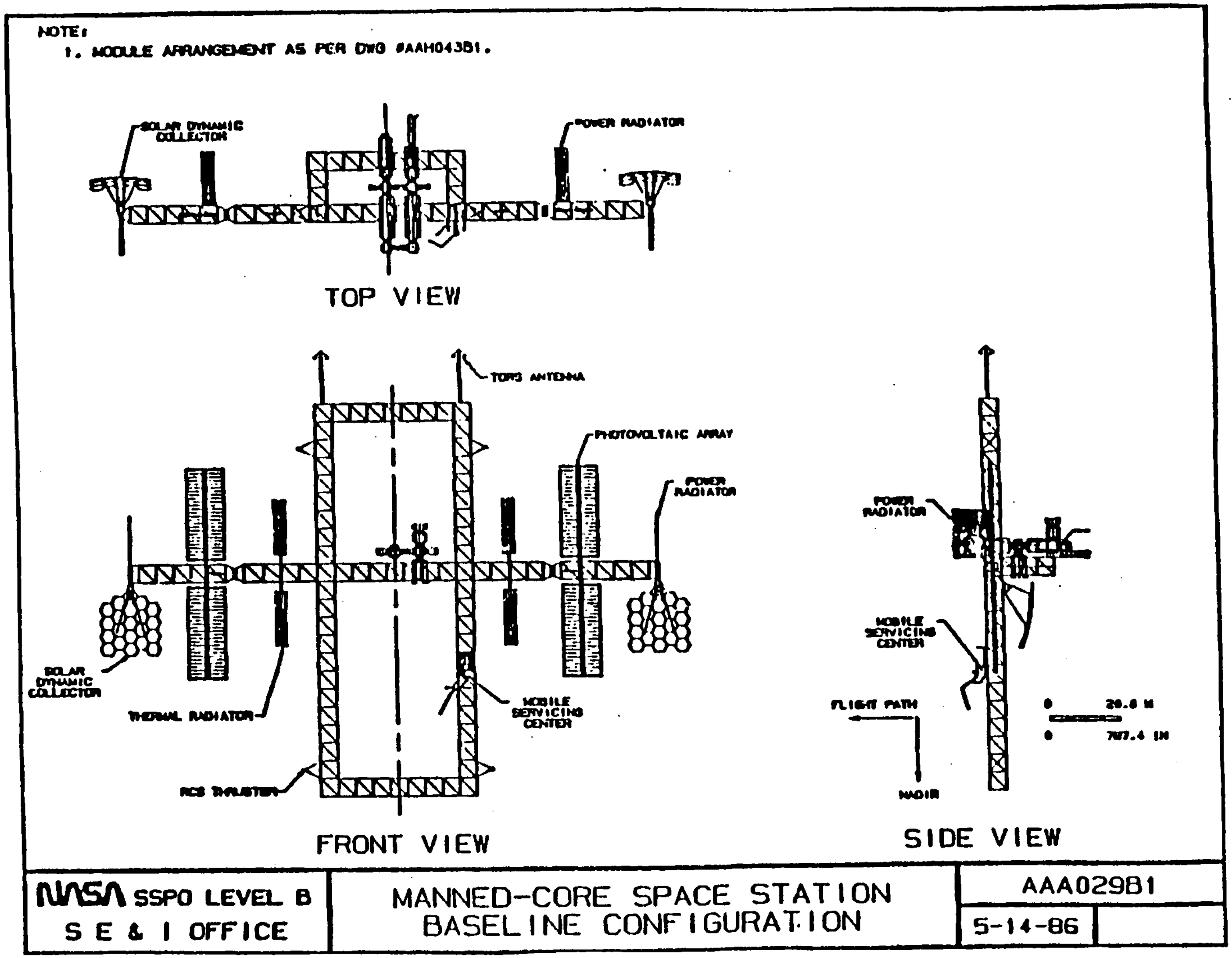


Figure 7.18-1. NASA Manned-Core Space Station Baseline Configuration.

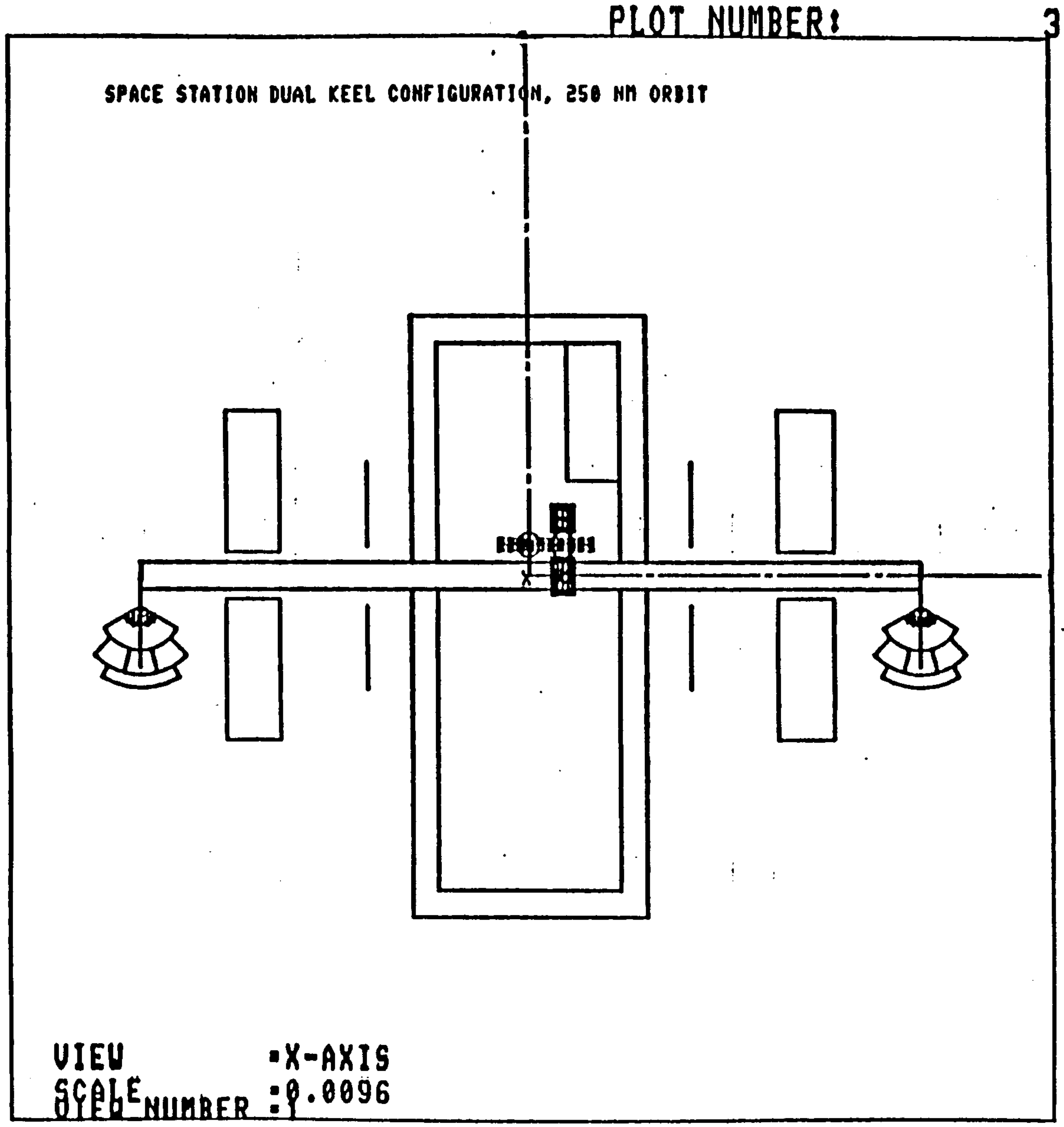


Figure 7.18-2. Computerized Model of Dual Keel Space Station

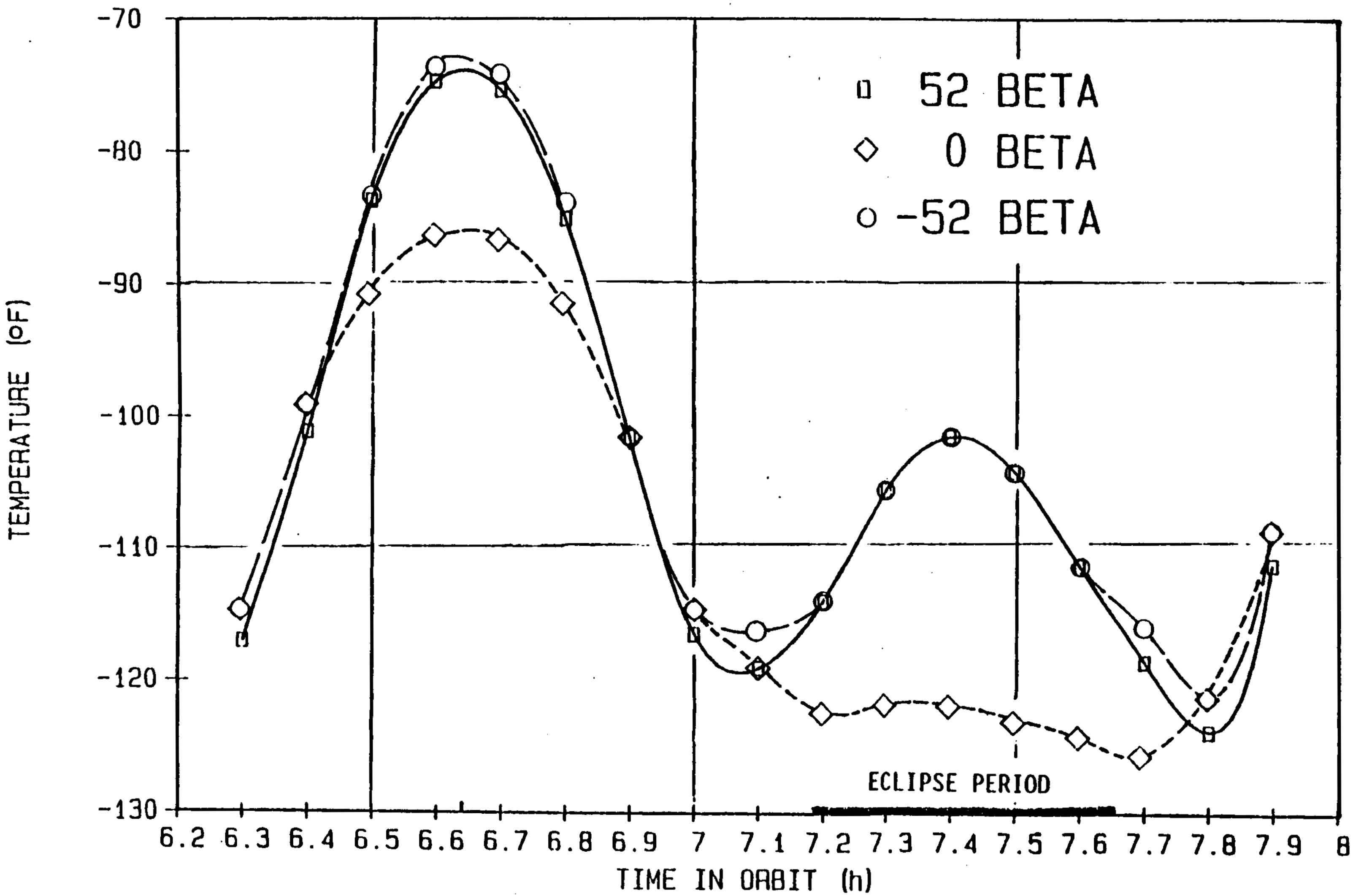


Figure 7.18-3. Orbital Variations in SD Radiator Sink Temperatures

7.19 PMAD Feeder Study

Design and evaluation of cable technologies which support 20 kHz primary and secondary distribution resulted in the following conclusions:

a. Cables used for primary distribution for distances between PDCAs should be a high surface area configuration, such as "Litz" type wire to minimize the added resistance and its attendant voltage drop due to skin effect at 20 kHz. The detail design of the cable depends on the permissible losses, current, and cable length. For example, to deliver 25 kW at 440 VAC to a load 150 meters distant, and limit the maximum losses to 2.0%, a cable would be made of approximately (2000) #34 AWG insulated strands. The analysis in support of these conclusions is presented in paragraph 7.19.1.

b. Cables used for secondary or short distance distribution (i.e., inside modules) can be Litz wire or conventional cables without adding significant system losses or cable mass.

c. Cable designs and capacities are driven by system considerations (such as allowable losses, voltage drops, reactance effects, EMI, etc.) and normal overload and rating concepts will not usually apply. For example, running twice the rated current in a particular bus will certainly result in four times the expected losses and in increased bus temperature, but not in bus failure or overstress.

d. The above considerations also create the need for controlled bus geometries.

   A woven Litz stripline developed by Induction General with LeRC has been built, tested, and characterized with real data and it has been selected as the baseline.

e. Because of a large number of unspecified concerns about distribution system instabilities and circulating currents due to cable reactances at 20 kHz, computer modeled stability and load center node performance analyses of possible station configurations,using the Induction General Stripline cable parameters have been performed. These have predicted that the system will be unconditionally stable and that bus voltages will always be within reasonable tolerances for loads connected at any point, and are supported by confirming test data from the General Dynamics test bed. A thorough discussion of that analysis and its detailed results are presented in paragraph 7.19.2.

7.19.1 Cable Loss Considerations

AC power transmission over long distances must consider added effective line resistance due to non-uniform current distribution in a solid wire (skin effect). At 20 kHz, the effect is large enough to at least double the resistance measured at DC. Therefore, the primary distribution cables between PDCAs should be some form of high surface area construction to minimize these AC effects.

Litz wire is widely available and commonly used in high frequency transformer design, and it was chosen for evaluation.

In order to bound the problem and to develop an understanding of the actual cable sizes involved, parametric data was developed for losses and cable masses from a reference cable with the following capabilities:

Power =	20.0 kW
Line Voltage =	440 VAC, RMS
Line Length =	160 meters; (320 meters, round trip)

Figure 2.3.5-2 shows the AC/DC resistance ratio for a 20 kHz cable as a function of individual Litz strand gauge for a family of resistances. It was obtained from solutions of the classical Litz wire resistance ratio equation developed by S. Butterworth and found in the "Radio Engineer's Handbook" by F. E. Terman. That equation is:

$$R_{ac}/R_{dc} = H + k(nd_s/d_o)^2G$$

where:

(d_s)	is the strand diameter
(d_o)	is the cable diameter
(n)	is the number of strands
(k)	$= 2.0$ for $n > 100$
(H)	is the individual strand resistance resistance ratio
(G)	is the proximity effect factor due to nearby strands

(H) and (G) are functions of the individual strand diameter, conductivity, permeability, and the AC frequency. They are calculated from the appropriate equations also found in Terman's Handbook.

The specific resistance parameters (150, 300, and 600 u /meter) are the values corresponding 0.5%, 1.0%, and 2.0% losses in the reference cable. This figure graphically illustrates the importance of small strand size as cable specific resistance or required losses are reduced.

Figure 7.19-1 shows cable specific weight as a function of strand size for the same specific resistances used in Figure 7.19-2. The slope of the low resistance curves shows that if strand sizes are chosen that are too large, acceptable losses will not be possible, even with unacceptable weights.

If we consider the point design of a typical Space Station cable, using the above data, and optimizing from the loss point of view only, it would result in the following:

<u>Requirements</u>	
Power =	25.0 kW
Voltage =	440 VAC, RMS
Length =	150 meters
Maximum Losses =	2.0%
<u>Resulting Configuration</u>	
Maximum Resistance =	0.155 ohms, (round trip) 516 microhms/meter
Maximum Strand Size =	#34 AWG

7.19.2 Cable Performance Analysis

Cable design for AC systems is driven by many interrelated requirements, most of which do not involve the maximum current carrying capacity of the wire. Since distributed inductance causes a voltage drop which is in quadrature with the load voltage, it can be reasonably large (approaching 10%) before its effect has an impact. However, an unshielded 44 volt end-to-end voltage drop on a 440 volt bus would provide an electrostatic field excitation which was totally unacceptable from an EMI point of view. Since inductance is also directly related to net magnetic flux linkages, its magnitude is also an indicator of stray magnetic fields present in the cable vicinity, another EMI concern. Therefore, we must try to reduce distributed inductance below those values dictated by the usual voltage regulation concerns.

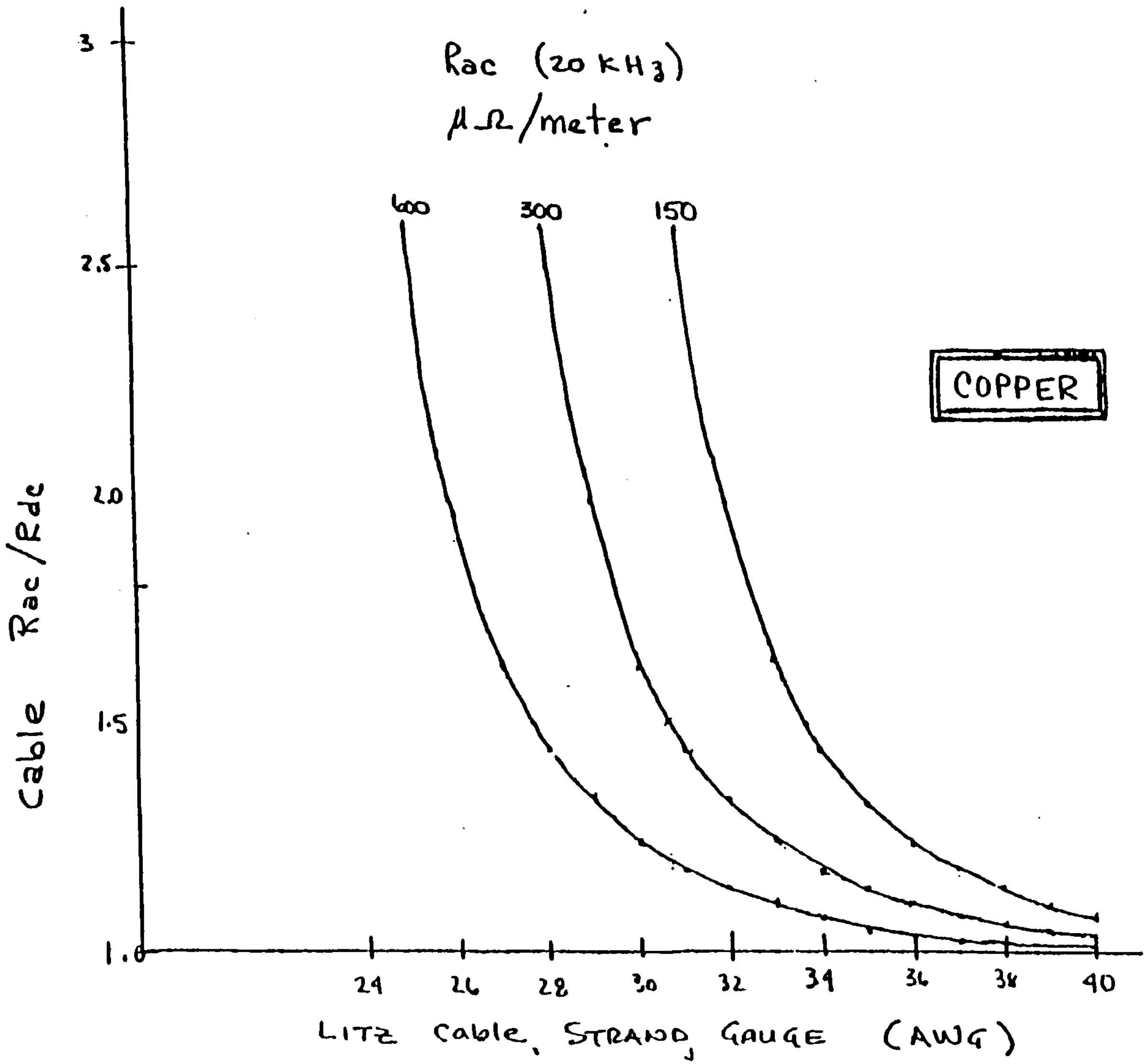


Figure 7.19-1 Cable Resistance Ratio versus Gauge

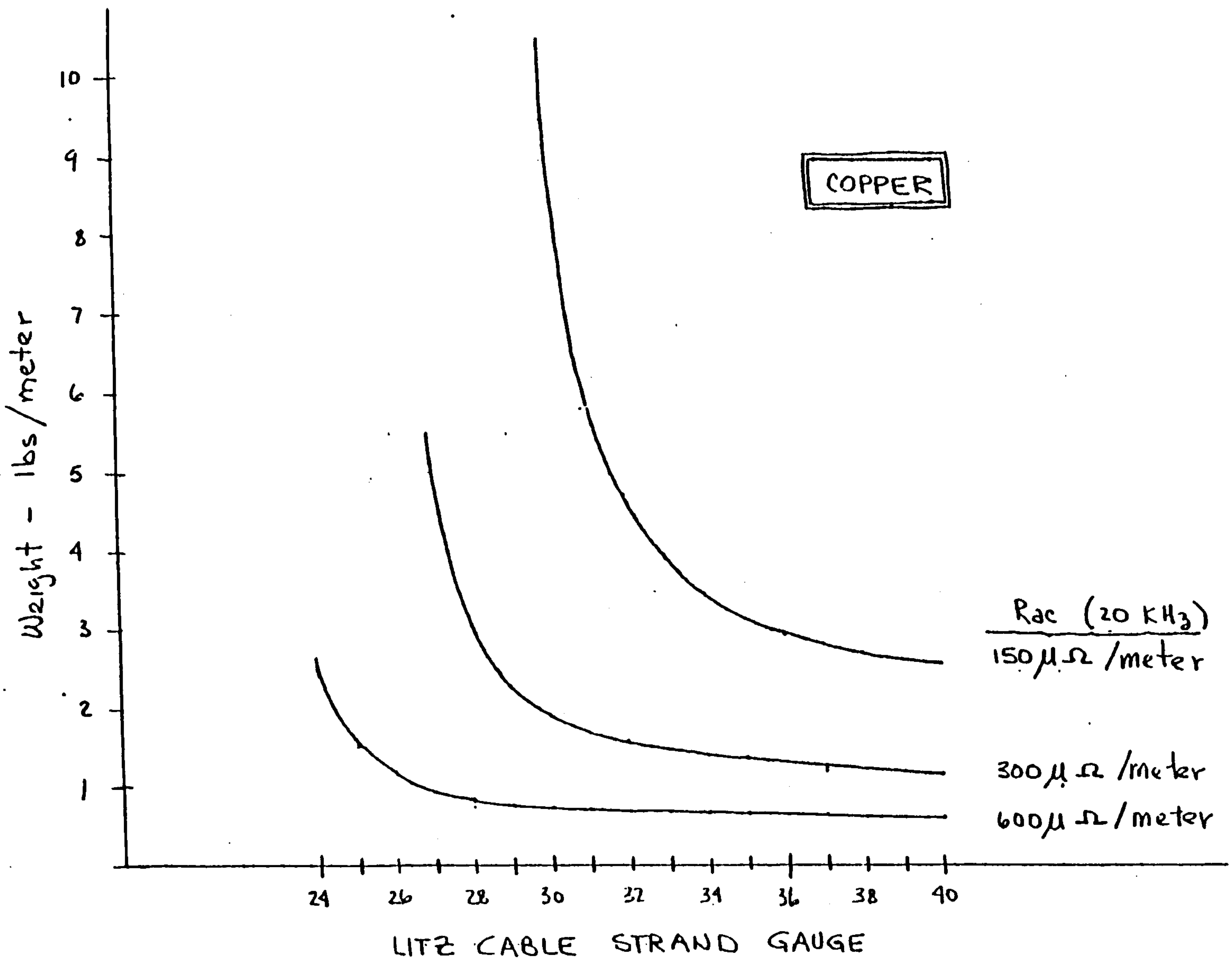


Figure 7.19-2 Cable Specific Weight versus Gauge

In general, if we reduce cable inductance, we pay a penalty in increased distributed capacitance. If capacitances in long cables are too high, high AC shunt currents flow in the system, which require additional reactive current capability from the source and cause additional I^2R losses for the system. Without going into the details of the original design trades used, we can say that the Induction General cable represents a point solution of the problem, appropriate to a nominal 25 kW bus in the Space Station configuration. Its distributed parameters (measured from configuration demonstration samples of typical Space Station lengths) are:

Resistance =	0.083 m-ohms/meter
Inductance =	0.035 u-henries/meter
Capacitance =	0.00137 u-farad/meter

Using this set of distributed parameters, the power bus was modeled for a generic branch/loop configuration as a function of length, and constructed, loaded and unloaded bus models for possible primary bus branches, resulting in three sizes.

Module Network =	(4) @ 100 meters; 25 kW
Upper Keel Network =	(2) @ 150 meters; 15 kW
Lower Keel Network =	(2) @ 170 meters; 15 kW

"Micro-cap" running on a MacIntosh(TM) computer was used as the primary analysis tool and selected points were confirmed and calibrated with hand analysis. System stability at 20 kHz was investigated by modeling the different transmission line lengths as a number of discrete sections and placing resistive and reactive loads at typical PDCA locations. The configurations examined were:

- o Unloaded bus
- o Bus loaded with its maximum rated load, located at 1/4, 1/2, and 3/4 of the distance along its length
- o Bus loaded with its maximum rated load, divided and distributed at the 1/4, 1/2, and 3/4 points simultaneously
- o (+) and (-) 0.9 power factor loads

The results indicated that for all cases the uncontrolled voltage along the line is within +1.4% and -0.7% of the nominal value, and less than 0.5% from nominal for near-unity power factor loads.

The same analysis was performed for the upper keel busses, loads, and PDCA locations. The same set of above listed conditions were applied with similar results; the uncontrolled line voltage is always within +1.6% and -1.4% of the nominal value, and less than 0.5% from nominal for near-unity power factor loads. Since the lower keel model has only minor differences, these conclusions are valid for it also.

Circulating reactive current for the line itself due to distributed shunt capacitance is 11.4 amp in the worst case, clearly not a problem for a line designed for approximately 60 amp and carrying 34.1 in the upper keel. The module busses would have 7.6 amp with a load current of 56.8 amp. Since the current limits are based on voltage drop and apportioning the system losses, these values must be considered in the overall system calculations.

7.20 PMAD DISTRIBUTION ARCHITECTURE TRADE STUDY

7.20.1 Introduction

This trade study report documents the work performed in evaluating the merits of various PMAD power distribution architectures.

7.20.2 Task Description

Evaluate the following PMAD generic architectures:

a) Ring
b) Radial
c) Network
d) Star

using a mathematical model of each and compare on the basis of mass, efficiency, protection methods, and switchgear requirements.

Evaluate distribution voltages of 440 and 208 on a similar basis.

7.20.3 Assumptions

In setting up the various architectures for evaluation, the following requirements were placed on each configuration:

(1) Loading at each location (bus) shall be identical (the bus loading values are discussed later).

(2) Acceptable failure tolerances shall be one failure tolerant for one half (side) of station PMAD.

(3) External MBSA and PDCA locations (10 places) shall be used for cable length parameters (see Figure 7.20-1).

(4) Cable weights, capacities, and electrical characteristics shall be based on NASA advanced development cable designed by Induction General (see Figure 7.20-2). Cables shall meet 2.5% voltage drop at most remote point in system under assumed worst case load applied by (1) above.

(5) Fault protection methods (for hard faults) shall be hardware implemented. Additional equipment requirements such as control cables if needed shall be identified.

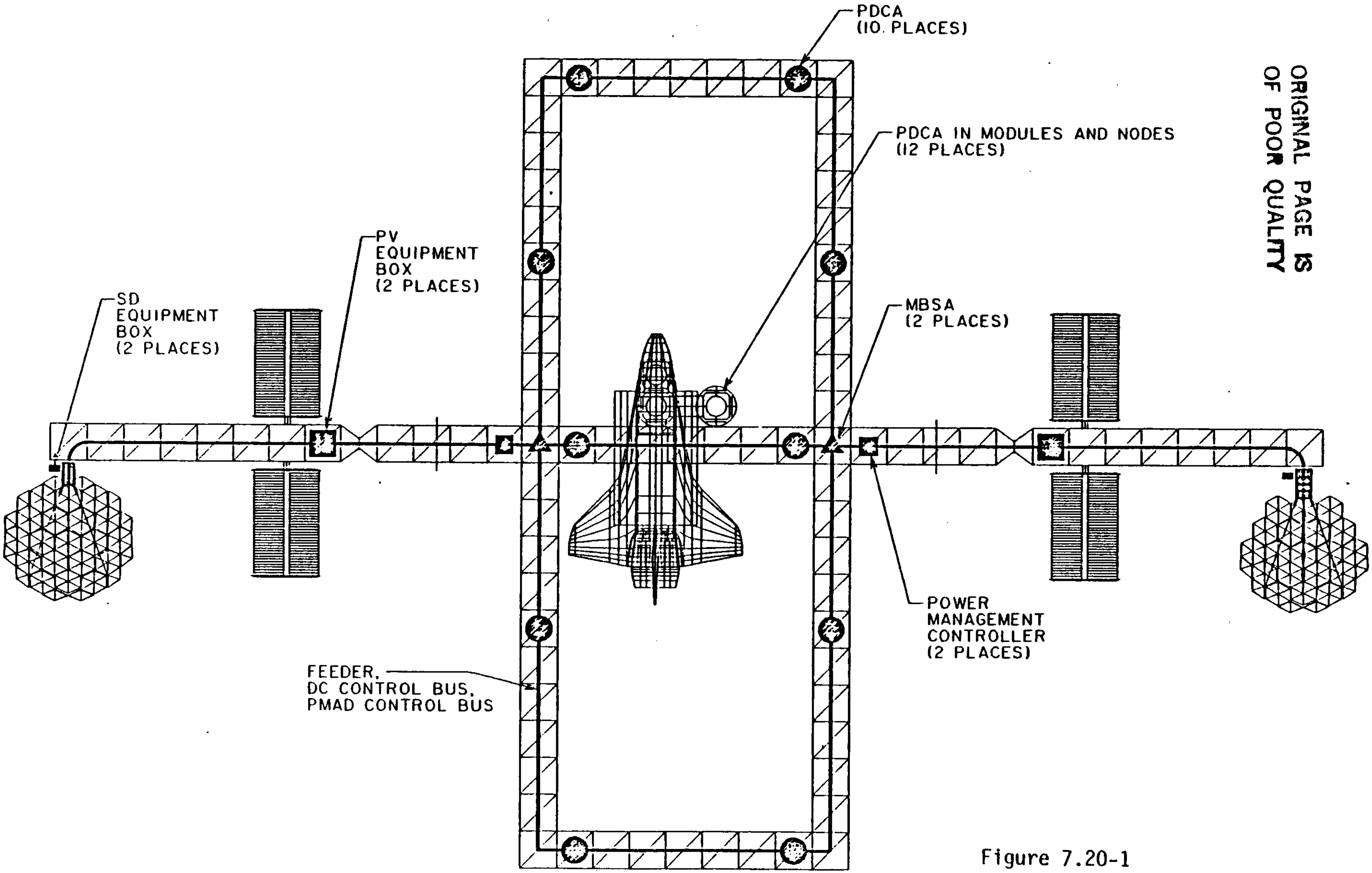


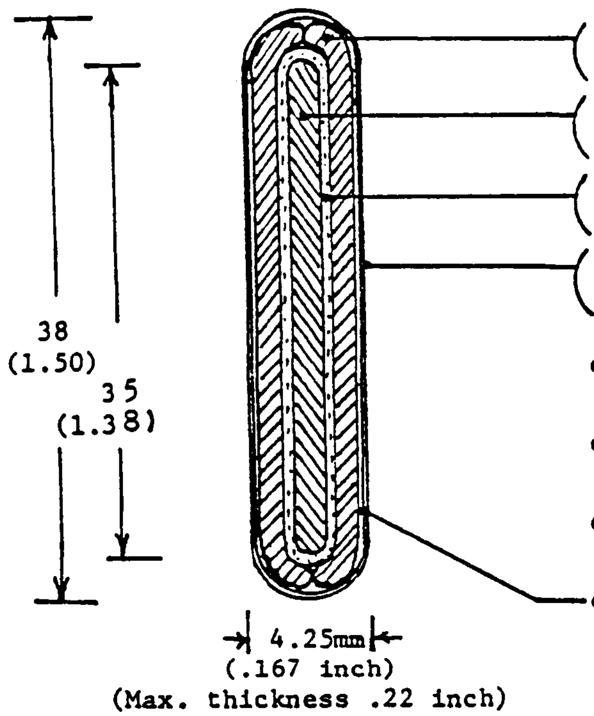
Figure 7.20-1

Rockwell International Corporation
Rocketdyne Division
Canoga Park, California

SPACE STATION EPS COMPONENT LOCATIONS

SIZE C | FSCM NO 02602 | DWG NO 7R070021
SCALE NONE | DO NOT SCALE PRINT | SHEET

DWG NO. 7R070021



- No. of strands in the middle strap=366 approximately
- No. of strands in each outer strap=183 approximately
- Strands=30 gage copper with solderable Nylese insulation
- One mil Teflon tape, half lap

Profax: Registered tradename of Hercules, Inc.

Nylese: Regsitered tradename of Philps Dodge Co.

Estimated Line Parameters

Resistance	2.256	mΩ/m	8.123 watts/m
Reactance	2.256	mΩ/m	.018 μH/m
Capacitance	0.003	μF/m	
Mass	0.372	Kg/m	

For 65-meter-long line operating at 440 Vrms, 60 Arms, 20 kHz.

Power Rating of the Line = 26.4 KVA

Power Losses at rated load = 2%

Voltage regulation at unity p.f. = 2.02%

Voltage regulation at zero p.f. lagging = 2.02%

Voltage regulation at .707 p.f. lagging = 2.83%

NASA ADVANCED DEVELOPMENT CABLE DESIGN
(INDUCTION GENERAL)

FIGURE 7.20-2

7.20.4 Model

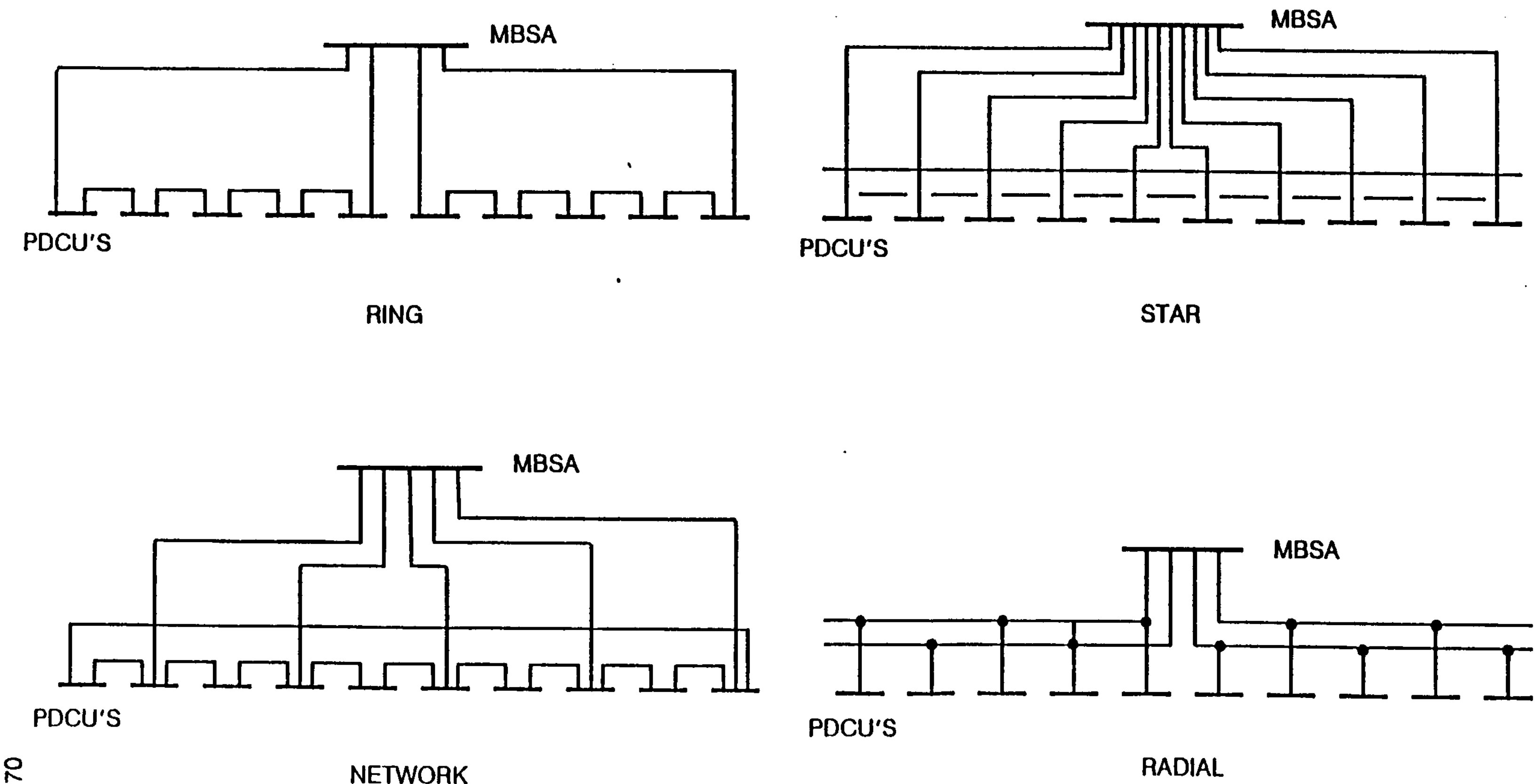
Figure 7.20-3 shows the various generic distribution architectures used for evaluation. The upper bus in each diagram is the MBSA on one side of the Space Station and connected to it are 10 load buses which would correspond to the 10 external PDCUs inside the alpha joint on the Space Station. The only difference between each architecture shown will be the method or configuration used to connect the PDCUs (load buses) to the MBSA (source bus). Using these various architectures, a generic network diagram, Figure 7.20-4, was developed. By assigning the proper impedances (Z) at each location, this generic network diagram can be used for all calculations. For instance, if a connection is not used in a particular architecture, then the impedance value would be set to infinity or some appropriately high value. Elsewhere, appropriate cable parameter values are used.

For load impedances, a nominal load bus voltage is assumed and used with the wattage and power factor of the load to obtain an impedance. The load values used were based on data distributed by NASA Level B on external power requirements and summarized in Table 7.20-1. A power factor of .9 lagging was assumed for all loads. A detailed loads analysis is also discussed on Section 7.22.

Since the distribution frequency is 20 kHz, the equivalent circuit assumed for the cable was that of a long transmission line (with respect to the wavelength) (see Stevenson, Elements of power system analysis, 4th Edition, page 106). The equivalent PI-circuit is shown in Figure 7.20-5 and was used in the model calculations. Capacitive charging currents are included.

A load flow program was then written that uses the complex impedances of the network and the voltage source data as inputs. Load bus voltage and phase angles are computed and then applied to the impedance values to determine line current flow. This process is repeated for each configuration and iterated as required to meet the assumptions previously noted.

GENERIC DISTRIBUTION ARCHITECTURES

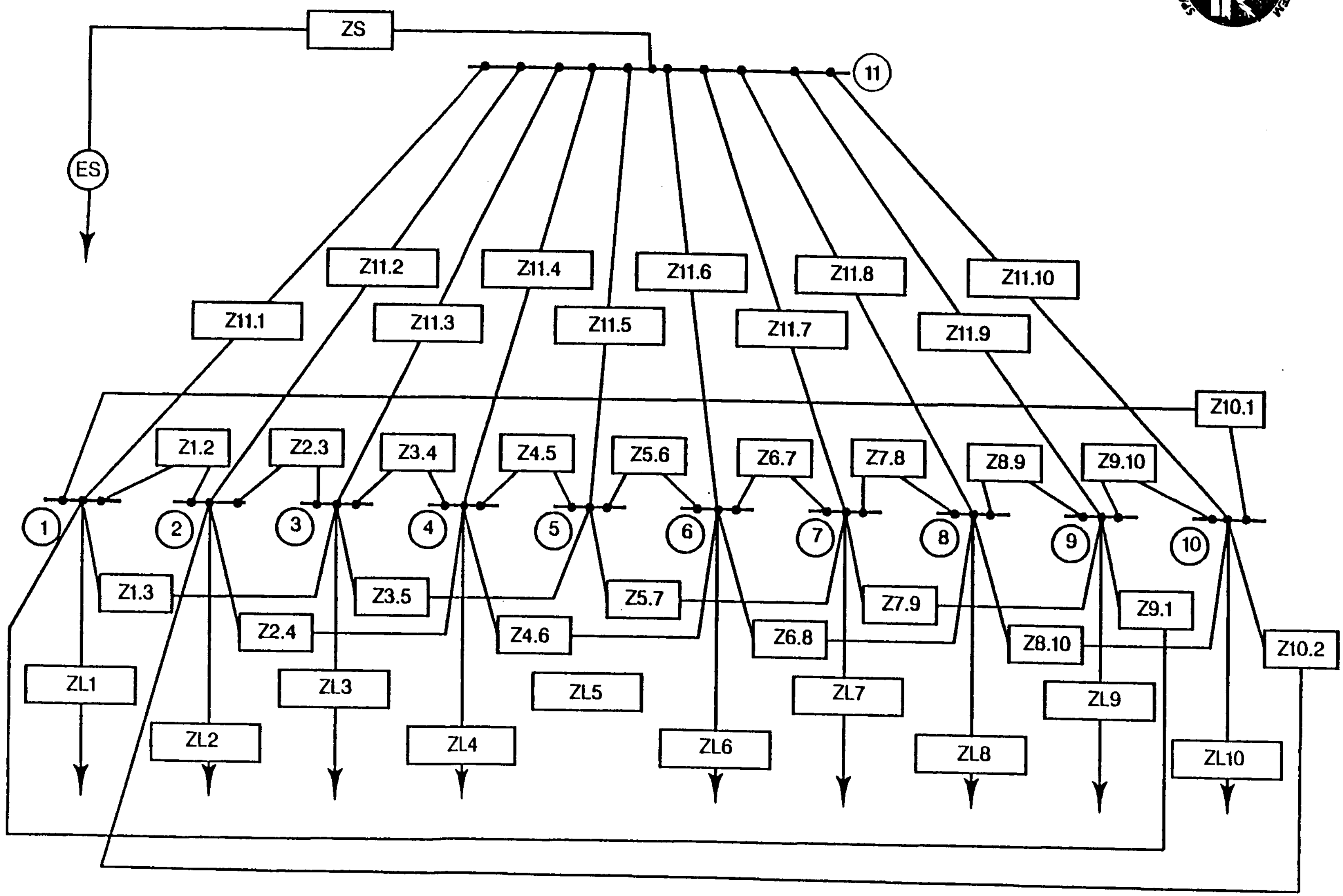


Q86D-13-1470

Rockwell International
Rocketdyne Division

Figure 7.20-3

GENERIC NETWORK DIAGRAM



Rockwell International
Rocketdyne Division

Figure 7.20-4

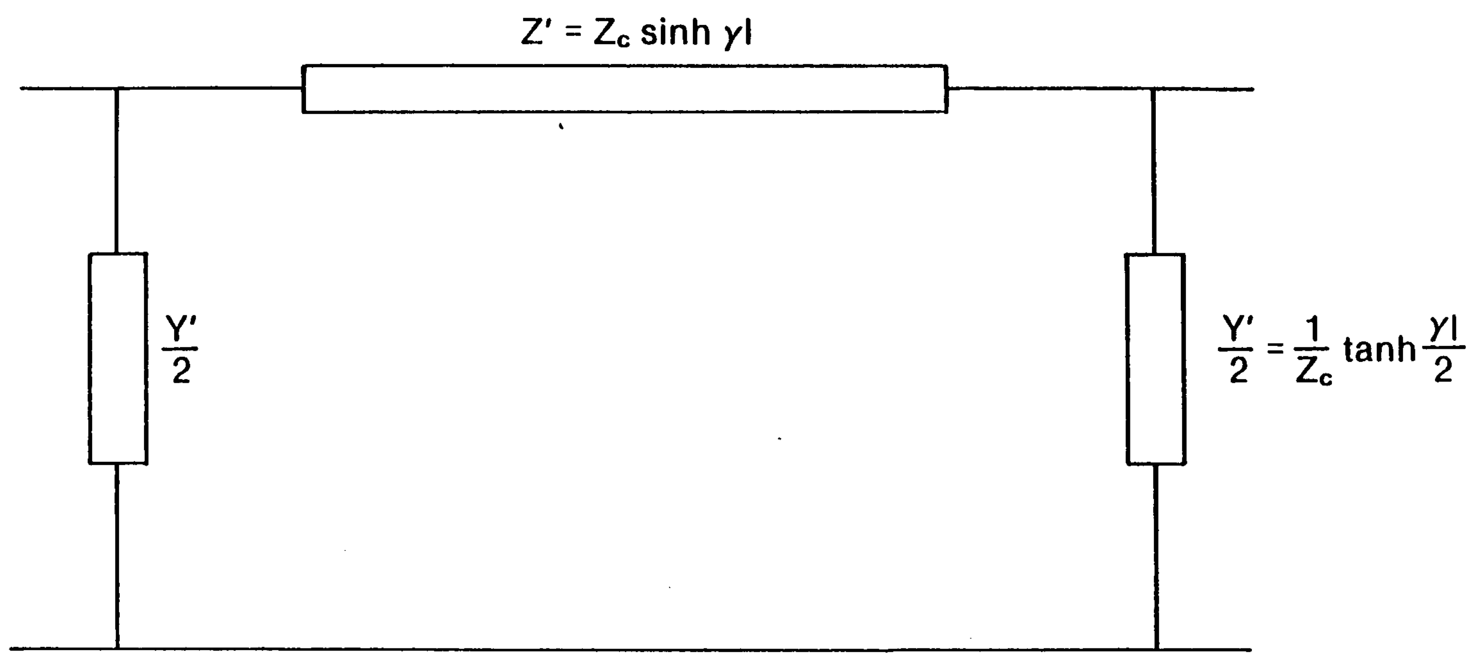


Figure 7.20-5
EQUIVALENT PI - CIRCUIT

TABLE 7.20-1. LOADS ANALYSIS

	Maximum Demand (kW)
LKS	25
LBS	16.9
LBP	16.9
LKP	17.5
TBP	14.9
UKP	22.3
UBP	19.3
UBS	19.3
UKS	13.6
TBS	14.9

TABLE 7.20-2. TRADE STUDY DATA SUMMARY

	RING	NETWORK	STAR	RADIAL
Max V	450	450	450	450
Min V	437	438	436	428
Max I	98	82	60	105
Watts Lost	1980	2100	1728	2294
Cable Mass	390	404	507	556
Max V	219	219	219	219
Min V	207	207	206	197
Max I	200	168	118	212
Watts Lost	4020	4298	3371	4547
Cable Mass	797	826	1038	1137

7.20.5 Results

Table 7.20-2 is a summary of the data generated by the trade study. (Detailed data is given in Table 7.20-3 through -10.) Using the 440 volt RING configuration as a baseline, it is observed that the STAR connection is the only configuration with improved efficiency with losses reduced 13%. However, the STAR cable mass is 30% higher than the RING. The cable mass associated with all other configurations is higher, the RADIAL being the highest with a 43% increase over the RING. On the basis of mass and efficiency, the NETWORK is an acceptable alternative to the RING since losses are increased only 6% and mass only 4%. The 208V distribution system is obviously much heavier and less efficient as expected due to the higher current flows.

RING

id#	I re	I im	I mag	I angle	id#
11.1	61.83	-23.82	66.26	-0.37	11.1
11.2					11.2
11.3					11.3
11.4					11.4
11.5	92.12	-31.95	97.51	-0.33	11.5
11.6	81.37	-29.78	86.65	-0.35	11.6
11.7					11.7
11.8					11.8
11.9					11.9
11.10	58.69	-19.65	61.89	-0.32	11.10
1.2	8.31	0.37	8.32	0.04	1.2
2.3	-17.25	8.27	19.13	-3.59	2.3
3.4	-42.91	15.72	45.70	-3.49	3.4
4.5	-70.17	23.77	74.09	-3.47	4.5
5.6					5.6
6.7	43.54	-13.66	45.63	-0.30	6.7
7.8	12.55	-3.23	12.96	-0.25	7.8
8.9	-18.36	7.23	19.73	-3.52	8.9
9.10	-37.05	11.52	38.80	-3.44	9.1
10.1					10.1
1.3					1.3
2.4					2.4
3.5					3.5
4.6					4.6
5.7					5.7
6.8					6.8
7.9					7.9
8.10					8.10
9.1					9.1
10.2					10.2

	Re	Im		Mag	Angle
v1	437.72	-6.54	v1	437.77	-0.01
v2	437.28	-7.12	v2	437.34	-0.02
v3	438.82	-6.42	v3	438.86	-0.01
v4	442.89	-4.06	v4	442.91	-0.01
v5	448.48	-0.64	v5	448.49	.00
v6	444.19	-3.10	v6	444.20	-0.01
v7	440.80	-5.28	v7	440.83	-0.01
v8	439.87	-5.95	v8	439.91	-0.01
v9	441.40	-5.11	v9	441.43	-0.01
v10	444.27	-3.25	v10	444.29	-0.01
v11	449.71	0.10	v11	449.71	.00

RING

MAX VOLTAGE MAG=	449.71	
MIN VOLTAGE MAG=	437.34	
MAX PHASE SHIFT=	0.93	
MAX CURRENT=	97.51	
WATTS LOST=	1980	
CABLE MASS(lbm)=	389.56	
VOLT SOURCE,p=	450	.00
FREQUENCY=	20000	

CABLE PARAMETERS

MAIN FEEDERS		TIE FEEDERS	
r1=	1.9E-03	r2=	1.9E-03
L1=	1.8E-08	L2=	1.8E-08
c1=	3.0E-09	c2=	3.0E-09
amp=	75	amp=	75
kg/m=	0.465	kg/m=	0.465

nasa cable

LOAD DATA					LOCATION	KW	P.F.	MAX
L1	53.65	-26.78	60.05	-0.47	LKS	25.00	0.90	25.00
L2	25.49	-12.86	28.56	-0.47	LBS	11.90	0.90	16.90
L3	25.60	-12.87	28.66	-0.47	LBP	11.90	0.90	16.90
L4	27.22	-13.49	30.38	-0.46	LKP	12.50	0.90	17.50
L5	21.91	-10.65	24.36	-0.45	TBP	9.90	0.90	14.90
L6	37.82	-18.64	42.17	-0.46	UKP	17.30	0.90	22.30
L7	30.95	-15.45	34.59	-0.46	UBP	14.30	0.90	19.30
L8	30.86	-15.46	34.52	-0.46	UBS	14.30	0.90	19.30
L9	18.64	-9.30	20.83	-0.46	UKS	8.60	0.90	13.60
L10	21.64	-10.68	24.13	-0.46	TBS	9.90	0.90	14.90
source	294.35	3.99	294.37	0.01	IMPEDANCE,r=		0.001	.00

ORIGINAL PAGE IS OF POOR QUALITY

LOAD ANALYSIS (RING, 440 VAC)

TABLE 7.20-3

ORIGINAL PAGE IS
OF POOR QUALITY

NETWORK

id#	I re	I im	I mag	I angle	id#
11.1					11.1
11.2	36.38	-11.72	38.22	-0.31	11.2
11.3					11.3
11.4	73.38	-21.85	77.04	-0.29	11.4
11.5					11.5
11.6	78.40	-24.57	82.16	-0.30	11.6
11.7					11.7
11.8	34.37	-10.70	36.00	-0.30	11.8
11.9					11.9
11.10	73.04	-22.53	76.45	-0.30	11.10
1.2	-18.07	7.51	19.57	-3.54	1.2
2.3	-7.52	3.17	8.16	-3.54	2.3
3.4	-33.53	10.32	35.08	-3.44	3.4
4.5	12.75	-3.73	13.29	-0.28	4.5
5.6	-9.06	1.70	9.21	-3.33	5.6
6.7	31.32	-9.49	32.73	-0.29	6.7
7.8	0.10	0.68	0.68	1.43	7.8
8.9	3.38	-0.09	3.38	-0.03	8.9
9.10	-15.49	3.97	15.99	-3.39	9.1
10.1	35.75	-13.50	38.21	-0.36	10.1
1.3					1.3
2.4					2.4
3.5					3.5
4.6					4.6
5.7					5.7
6.8					6.8
7.9					7.9
8.10					8.10
9.1					9.1
10.2					10.2

	Re	Im		Mag	Angle
v1	438.01	-5.60	v1	438.05	-0.01
v2	439.85	-4.92	v2	439.87	-0.01
v3	440.61	-4.65	v3	440.64	-0.01
v4	444.30	-2.87	v4	444.31	-0.01
v5	443.11	-3.46	v5	443.12	-0.01
v6	443.89	-2.97	v6	443.90	-0.01
v7	440.95	-4.41	v7	440.97	-0.01
v8	440.99	-4.46	v8	441.01	-0.01
v9	440.73	-4.68	v9	440.76	-0.01
v10	442.14	-3.92	v10	442.16	-0.01
v11	449.70	0.07	v11	449.70	.00

NETWORK
MAX VOLTAGE MAG= 449.70
MIN VOLTAGE MAG= 438.05
MAX PHASE SHIFT= 0.73
MAX CURRENT= 82.16
WATTS LOST= 2100
CABLE MASS(lbm)= 403.91
VOLT SOURCE,p= 450 .00
FREQUENCY= 20000

CABLE PARAMETERS

MAIN FEEDERS		TIE FEEDERS	
r1=	2.3E-03	r2=	2.6E-03
L1=	1.8E-08	L2=	1.8E-08
c1=	3.0E-09	c2=	3.0E-09
amp=	60	amp=	45
kg/m=	0.372	kg/m=	0.279

nasa cable

LOAD DATA					LOCATION	KW	P.F.	MAX
L1	53.74	-26.88	60.09	-0.46	LKS	25.00	0.90	25.00
L2	25.71	-12.81	28.72	-0.46	LBS	11.90	0.90	16.90
L3	25.76	-12.81	28.77	-0.46	LBP	11.90	0.90	16.90
L4	27.34	-13.46	30.47	-0.46	LKP	12.50	0.90	17.50
L5	21.58	-10.66	24.07	-0.46	TBP	9.90	0.90	14.90
L6	37.80	-18.62	42.14	-0.46	UKP	17.30	0.90	22.30
L7	30.99	-15.39	34.60	-0.46	UBP	14.30	0.90	19.30
L8	30.99	-15.40	34.60	-0.46	UBS	14.30	0.90	19.30
L9	18.62	-9.26	20.80	-0.46	UKS	8.60	0.90	13.60
L10	21.52	-10.66	24.02	-0.46	TBS	9.90	0.90	14.90
source	296.58	4.04	296.60	0.01	IMPEDANCE,r=		0.001	.00

LOAD ANALYSIS (NETWORK, 440 VAC)

TABLE 7.20-4

STAR

id#	I re	I im	I mag	I angle	id#
11.1	53.48	-26.11	59.51	-0.45	11.1
11.2	25.78	-12.38	28.60	-0.45	11.2
11.3	26.01	-12.58	28.39	-0.45	11.3
11.4	27.58	-13.36	30.65	-0.45	11.4
11.5	21.97	-10.64	24.41	-0.45	11.5
11.6	38.07	-18.50	42.33	-0.45	11.6
11.7	31.22	-15.14	34.69	-0.45	11.7
11.8	30.90	-14.90	34.30	-0.45	11.8
11.9	18.85	-9.09	20.93	-0.45	11.9
11.10	21.82	-10.57	24.25	-0.45	11.10
1.2			.00		1.2
2.3			.00		2.3
3.4			.00		3.4
4.5			.00		4.5
5.6			.00		5.6
6.7			.00		6.7
7.8			.00		7.8
8.9			.00		8.9
9.10			.00		9.1
10.1			.00		10.1
1.3					1.3
2.4					2.4
3.5					3.5
4.6					4.6
5.7					5.7
6.8					6.8
7.9					7.9
8.10					8.10
9.1					9.1
10.2					10.2

	Re	Im		Mag	Angle
v1	435.66	-3.51	v1	435.68	-0.01
v2	441.02	-2.17	v2	441.02	.00
v3	443.87	-1.43	v3	443.88	.00
v4	447.13	-0.57	v4	447.13	.00
v5	449.29	.00	v5	449.29	.00
v6	446.15	-0.83	v6	446.15	.00
v7	443.29	-1.58	v7	443.29	.00
v8	439.87	-2.46	v8	439.88	-0.01
v9	445.13	-1.10	v9	445.13	.00
v10	446.86	-0.65	v10	446.86	.00
v11	449.70	0.11	v11	449.70	.00

STAR
MAX VOLTAGE MAG= 449.70
MIN VOLTAGE MAG= 435.68
MAX PHASE SHIFT= 0.46
MAX CURRENT= 59.51
WATTS LOST= 1728
CABLE MASS(lbm)= 507.45
VOLT SOURCE,p= 450 .00
FREQUENCY= 20000

CABLE PARAMETERS

MAIN FEEDERS		TIE FEEDERS	
r1=	2.6E-03	r2=	2.6E-03
L1=	1.8E-08	L2=	1.8E-08
c1=	3.0E-09	c2=	3.0E-09
amp=	45	amp=	45
kg/m=	0.279	kg/m=	0.279

nasa cable

LOAD DATA					LOCATION	KW	P.F.	MAX
L1	53.58	-26.48	59.76	-0.46	LKS	25.00	0.90	25.00
L2	25.86	-12.68	28.80	-0.46	LBS	11.90	0.90	16.90
L3	26.04	-12.72	28.98	-0.45	LBP	11.90	0.90	16.90
L4	27.58	-13.40	30.67	-0.45	LKP	12.50	0.90	17.50
L5	21.97	-10.64	24.41	-0.45	TBP	9.90	0.90	14.90
L6	38.08	-18.53	42.35	-0.45	UKP	17.30	0.90	22.30
L7	31.25	-15.27	34.78	-0.45	UBP	14.30	0.90	19.30
L8	30.98	-15.22	34.51	-0.46	UBS	14.30	0.90	19.30
L9	18.88	-9.20	21.00	-0.45	UKS	8.60	0.90	13.60
L10	21.83	-10.61	24.27	-0.45	TBS	9.90	0.90	14.90
source	296.15	3.97	296.18	0.01	IMPEDANCE,r=		0.001	.00

LOAD ANALYSIS (STAR, 440 VAC)

TABLE 7.20-5

ORIGINAL PAGE IS
OF POOR QUALITY

RADIAL

id#	I re	I im	I mag	I angle	id#
11.1					11.1
11.2					11.2
11.3					11.3
11.4	54.06	-15.09	56.13	-0.27	11.4
11.5	100.79	-29.63	105.06	-0.29	11.5
11.6	90.89	-24.71	94.18	-0.27	11.6
11.7	50.34	-14.39	52.36	-0.28	11.7
11.8					11.8
11.9					11.9
11.10					11.10
1.2					1.2
2.3					2.3
3.4					3.4
4.5					4.5
5.6					5.6
6.7					6.7
7.8					7.8
8.9					8.9
9.1					9.1
10.1					10.1
1.3	-52.29	22.16	56.79	-3.54	1.3
2.4	-26.16	7.18	27.13	-3.41	2.4
3.5	-78.36	24.36	82.06	-3.44	3.5
4.6					4.6
5.7					5.7
6.8	52.62	-11.05	53.77	-0.21	6.8
7.9	18.90	-4.14	19.35	-0.22	7.9
8.10	21.34	-5.73	22.10	-0.26	8.10
9.1					9.1
10.2					10.2

	Re	Im		Mag	Angle
v1	427.98	-14.56	v1	428.22	-0.03
v2	442.85	-5.47	v2	442.88	-0.01
v3	436.34	-9.70	v3	436.44	-0.02
v4	446.91	-2.41	v4	446.81	-0.01
v5	448.61	-0.84	v5	448.61	.00
v6	444.87	-4.13	v6	444.89	-0.01
v7	443.71	-4.99	v7	443.74	-0.01
v8	437.99	-10.16	v8	438.11	-0.02
v9	441.22	-7.14	v9	441.28	-0.02
v10	435.03	-12.48	v10	435.21	-0.03
v11	449.70	0.08	v11	449.70	.00

RADIAL
MAX VOLTAGE MAG= 449.70
MIN VOLTAGE MAG= 428.22
MAX PHASE SHIFT= 1.95
MAX CURRENT= 105.06
WATTS LOST= 2294
CABLE MASS(lbm)= 555.79
VOLT SOURCE,p= 450 .00
FREQUENCY= 20000

CABLE PARAMETERS

MAIN FEEDERS		TIE FEEDERS	
r1=	1.5E-03	r2=	1.9E-03
L1=	1.8E-08	L2=	1.8E-08
c1=	3.0E-09	c2=	3.0E-09
aeq=	90	aeq=	75
kg/m=	0.598	kg/m=	0.465

nasa cable

LOAD DATA					LOCATION	KW	P.F.	MAX
L1	51.97	-27.39	58.74	-0.49	LKS	25.00	0.90	25.00
L2	25.87	-12.93	28.92	-0.46	LBS	11.90	0.90	16.90
L3	25.37	-12.99	28.50	-0.47	LBP	11.90	0.90	16.90
L4	27.51	-13.51	30.65	-0.46	LKP	12.50	0.90	17.50
L5	21.91	-10.66	24.37	-0.45	TBP	9.90	0.90	14.90
L6	37.84	-18.76	42.23	-0.46	UKP	17.30	0.90	22.30
L7	31.16	-15.53	34.82	-0.46	UBP	14.30	0.90	19.30
L8	30.58	-15.70	34.38	-0.47	UBS	14.30	0.90	19.30
L9	18.59	-9.39	20.82	-0.47	UKS	8.60	0.90	13.60
L10	20.97	-10.91	23.64	-0.48	TBS	9.90	0.90	14.90
source	296.32	4.03	296.35	0.01	IMPEDANCE,r=		0.001	.00

LOAD ANALYSIS (RADIAL, 440 VAC)

TABLE 7.20-6

RING

id#	I re	I im	I mag	I angle	id#
11.1	121.89	-55.68	134.01	-0.43	11.1
11.2					11.2
11.3					11.3
11.4					11.4
11.5	183.44	-79.42	199.89	-0.41	11.5
11.6	162.26	-71.65	177.38	-0.42	11.6
11.7					11.7
11.8					11.8
11.9					11.9
11.10	116.94	-49.87	127.13	-0.40	11.10
1.2	16.51	-4.01	16.99	-0.24	1.2
2.3	-33.69	17.24	37.94	-3.61	2.3
3.4	-84.31	38.05	92.49	-3.57	3.4
4.5	-138.79	60.12	151.25	-3.55	4.5
5.6					5.6
6.7	86.31	-36.23	93.60	-0.40	6.7
7.8	24.79	-9.77	26.65	-0.38	7.8
8.9	-36.35	16.73	40.02	-3.57	8.9
9.10	-73.50	30.71	79.66	-3.54	9.1
10.1					10.1
1.3					1.3
2.4					2.4
3.5					3.5
4.6					4.6
5.7					5.7
6.8					6.8
7.9					7.9
8.10					8.10
9.1					9.1
10.2					10.2

	Re	Im		Mag	Angle
v1	207.19	-5.58	v1	207.26	-0.03
v2	206.60	-6.02	v2	206.69	-0.03
v3	208.10	-5.38	v3	208.17	-0.03
v4	212.30	-3.34	v4	212.32	-0.02
v5	218.13	-0.40	v5	218.13	.00
v6	213.70	-2.59	v6	213.71	-0.01
v7	210.11	-4.45	v7	210.15	-0.02
v8	209.10	-5.00	v8	209.16	-0.02
v9	210.66	-4.26	v9	210.70	-0.02
v10	213.71	-2.67	v10	213.73	-0.01
v11	219.41	0.25	v11	219.41	.00

RING
MAX VOLTAGE MAG= 219.41
MIN VOLTAGE MAG= 206.69
MAX PHASE SHIFT= 1.67
MAX CURRENT= 199.89
WATTS LOST= 4020
CABLE MASS(lbm)= 796.82
VOLT SOURCE,p= 220 .00
FREQUENCY= 20000

CABLE PARAMETERS

MAIN FEEDERS		TIE FEEDERS	
r1=	1.9E-03	r2=	1.9E-03
L1=	1.8E-08	L2=	1.8E-08
c1=	3.0E-09	c2=	3.0E-09
amp=	75	amp=	75
kg/m=	0.465	kg/m=	0.465

nasa cable

LOAD DATA					LOCATION	KW	P.F.	MAX
L1	105.62	-54.71	118.95	-0.48	LKS	25.00	0.90	25.00
L2	50.08	-26.08	56.46	-0.48	LBS	11.90	0.90	16.90
L3	50.53	-26.10	56.87	-0.48	LBP	11.90	0.90	16.90
L4	54.41	-27.42	60.93	-0.47	LKP	12.50	0.90	17.50
L5	44.58	-21.69	49.58	-0.45	TBP	9.90	0.90	14.90
L6	75.94	-37.92	84.88	-0.46	UKP	17.30	0.90	22.30
L7	61.44	-31.38	68.99	-0.47	UBP	14.30	0.90	19.30
L8	61.06	-31.40	68.66	-0.47	UBS	14.30	0.90	19.30
L9	37.06	-18.88	41.60	-0.47	UKS	8.60	0.90	13.60
L10	43.45	-21.72	48.57	-0.46	TBS	9.90	0.90	14.90

source 585.08 0.49 585.08 .00 IMPEDANCE,r= 0.001 .00

LOAD ANALYSIS (RING, 208 VAC)

TABLE 7.20-7

ORIGINAL PAGE IS
OF POOR QUALITY

ORIGINAL PAGE IS OF POOR QUALITY

NETWORK

id#	I re	I im	I mag	I angle	id#
11.1					11.1
11.2	71.99	-30.30	78.11	-0.40	11.2
11.3					11.3
11.4	146.85	-58.99	158.22	-0.38	11.4
11.5					11.5
11.6	156.01	-64.02	168.63	-0.39	11.6
11.7					11.7
11.8	68.14	-28.15	73.72	-0.39	11.8
11.9					11.9
11.10	144.61	-59.60	156.41	-0.39	11.10
1.2	-35.58	16.81	39.35	-3.58	1.2
2.3	-14.75	6.67	16.19	-3.57	2.3
3.4	-66.20	27.00	71.49	-3.53	3.4
4.5	25.46	-10.12	27.40	-0.38	4.5
5.6	-17.98	6.31	19.06	-3.48	5.6
6.7	61.94	-25.10	66.83	-0.39	6.7
7.8	0.05	0.88	0.88	1.51	7.8
8.9	6.52	-1.79	6.76	-0.27	8.9
9.10	-30.75	11.83	32.95	-3.51	9.1
10.1	70.57	-31.70	77.36	-0.42	10.1
1.3					1.3
2.4					2.4
3.5					3.5
4.6					4.6
5.7					5.7
6.8					6.8
7.9					7.9
8.10					8.10
9.1					9.1
10.2					10.2

	Re	Im		Mag	Angle
v1	207.33	-4.48	v1	207.38	-0.02
v2	209.17	-3.90	v2	209.20	-0.02
v3	209.92	-3.65	v3	209.95	-0.02
v4	213.73	-2.21	v4	213.74	-0.01
v5	212.48	-2.70	v5	212.50	-0.01
v6	213.34	-2.32	v6	213.35	-0.01
v7	210.28	-3.48	v7	210.31	-0.02
v8	210.31	-3.52	v8	210.34	-0.02
v9	210.02	-3.67	v9	210.05	-0.02
v10	211.51	-3.07	v10	211.53	-0.01
v11	219.41	0.22	v11	219.41	.00

NETWORK

MAX VOLTAGE MAG= 219.41
MIN VOLTAGE MAG= 207.38
MAX PHASE SHIFT= 1.24
MAX CURRENT= 168.63
WATTS LOST= 4298
CABLE MASS(lbm)= 826.17
VOLT SOURCE,p= 220 .00
FREQUENCY= 20000

CABLE PARAMETERS

MAIN FEEDERS		TIE FEEDERS	
r1=	2.3E-03	r2=	2.6E-03
L1=	1.8E-08	L2=	1.8E-08
c1=	3.0E-09	c2=	3.0E-09
amp=	60	amp=	45
kg/m=	0.372	kg/m=	0.279

nasa cable

LOAD DATA					LOCATION	KW	P.F.	MAX
L1	105.97	-54.18	119.02	-0.47	LKS	25.00	0.90	25.00
L2	50.96	-25.87	57.15	-0.47	LBS	11.90	0.90	16.90
L3	51.18	-25.89	57.36	-0.47	LBP	11.90	0.90	16.90
L4	54.92	-27.30	61.34	-0.46	LKP	12.50	0.90	17.50
L5	43.20	-21.60	48.30	-0.46	TBP	9.90	0.90	14.90
L6	75.85	-37.76	84.73	-0.46	UKP	17.30	0.90	22.30
L7	61.63	-31.12	69.04	-0.47	UBP	14.30	0.90	19.30
L8	61.64	-31.13	69.05	-0.47	UBS	14.30	0.90	19.30
L9	37.00	-18.72	41.47	-0.47	UKS	8.60	0.90	13.60
L10	42.96	-21.58	48.08	-0.47	TBS	9.90	0.90	14.90

source 588.41 0.60 588.41 .00 IMPEDANCE,r= 0.001 .00

LOAD ANALYSIS (NETWORK, 208 VAC)

TABLE 7.20-8

STAR

id#	I re	I im	I mag	I angle	id#
11.1	105.34	-52.38	117.64	-0.46	11.1
11.2	51.47	-25.00	57.22	-0.45	11.2
11.3	52.32	-25.48	58.19	-0.45	11.3
11.4	55.94	-27.18	62.20	-0.45	11.4
11.5	44.81	-21.66	49.77	-0.45	11.5
11.6	77.03	-37.54	85.70	-0.45	11.6
11.7	62.69	-30.63	69.78	-0.45	11.7
11.8	61.51	-30.04	68.45	-0.45	11.8
11.9	38.04	-18.43	42.27	-0.45	11.9
11.10	44.23	-21.48	49.17	-0.45	11.10
1.2			.00		1.2
2.3			.00		2.3
3.4			.00		3.4
4.5			.00		4.5
5.6			.00		5.6
6.7			.00		6.7
7.8			.00		7.8
8.9			.00		8.9
9.10			.00		9.1
10.1			.00		10.1
1.3					1.3
2.4					2.4
3.5					3.5
4.6					4.6
5.7					5.7
6.8					6.8
7.9					7.9
8.10					8.10
9.1					9.1
10.2					10.2

	Re	Im		Mag	Angle
v1	205.81	-3.15	v1	205.84	-0.02
v2	210.90	-1.94	v2	210.91	-0.01
v3	213.66	-1.24	v3	213.67	-0.01
v4	216.86	-0.42	v4	216.86	.00
v5	219.00	0.14	v5	219.00	.00
v6	215.89	-0.67	v6	215.89	.00
v7	213.09	-1.38	v7	213.10	-0.01
v8	209.80	-2.20	v8	209.81	-0.01
v9	214.89	-0.93	v9	214.89	.00
v10	216.58	-0.49	v10	216.58	.00
v11	219.41	0.25	v11	219.41	.00

STAR
MAX VOLTAGE MAG= 219.41
MIN VOLTAGE MAG= 205.84
MAX PHASE SHIFT= 0.38
MAX CURRENT= 117.64
WATTS LOST= 3371
CABLE MASS(lbm)=1037.96
VOLT SOURCE,p= 220 .00
FREQUENCY= 20000

CABLE PARAMETERS

MAIN FEEDERS		TIE FEEDERS	
r1=	2.6E-03	r2=	2.6E-03
L1=	1.8E-08	L2=	1.8E-08
c1=	3.0E-09	c2=	3.0E-09
amp=	45	amp=	45
kg/m=	0.279	kg/m=	0.279

nasa cable

LOAD DATA					LOCATION	KW	P.F.	MAX
L1	105.52	-53.11	118.13	-0.47	LKS	25.00	0.90	25.00
L2	51.62	-25.59	57.62	-0.46	LBS	11.90	0.90	16.90
L3	52.39	-25.75	58.37	-0.46	LBP	11.90	0.90	16.90
L4	55.95	-27.23	62.23	-0.45	LKP	12.50	0.90	17.50
L5	44.81	-21.67	49.77	-0.45	TBP	9.90	0.90	14.90
L6	77.05	-37.61	85.74	-0.45	UKP	17.30	0.90	22.30
L7	62.76	-30.90	69.96	-0.46	UBP	14.30	0.90	19.30
L8	61.67	-30.67	68.88	-0.46	UBS	14.30	0.90	19.30
L9	39.10	-18.66	42.42	-0.46	UKS	8.60	0.90	13.60
L10	44.25	-21.56	49.22	-0.45	TBS	9.90	0.90	14.90
source	594.21	0.46	594.21	.00	IMPEDANCE,r=		0.001	.00

LOAD ANALYSIS (STAR, 208 VAC)

TABLE 7.20-9

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RADIAL

id#	I re	I im	I mag	I angle	id#
11.1					11.1
11.2					11.2
11.3					11.3
11.4	107.89	-42.72	116.04	-0.38	11.4
11.5	194.78	-82.82	211.66	-0.40	11.5
11.6	178.07	-72.39	192.23	-0.39	11.6
11.7	99.72	-40.41	107.59	-0.39	11.7
11.8					11.8
11.9					11.9
11.10					11.10
1.2					1.2
2.3					2.3
3.4					3.4
4.5					4.5
5.6					5.6
6.7					6.7
7.8					7.8
8.9					8.9
9.1					9.1
10.1					10.1
1.3	-99.19	50.28	111.21	-3.61	1.3
2.4	-51.85	20.61	55.80	-3.52	2.4
3.5	-149.54	66.09	163.50	-3.56	3.5
4.6					4.6
5.7					5.7
6.8	101.62	-38.99	108.84	-0.37	6.8
7.9	37.15	-13.96	39.69	-0.36	7.9
8.10	41.00	-17.16	44.45	-0.40	8.10
9.1					9.1
10.2					10.2

	Re	Im		Mag	Angle
v1	197.08	-11.84	v1	197.43	-0.06
v2	211.95	-4.59	v2	212.00	-0.02
v3	205.38	-7.75	v3	205.52	-0.04
v4	216.24	-1.96	v4	216.25	-0.01
v5	218.24	-0.54	v5	218.24	.00
v6	214.13	-3.36	v6	214.15	-0.02
v7	212.89	-4.19	v7	212.93	-0.02
v8	206.46	-8.17	v8	206.62	-0.04
v9	210.11	-5.97	v9	210.19	-0.03
v10	203.27	-10.04	v10	203.52	-0.05
v11	219.42	0.23	v11	219.42	.00

RADIAL
MAX VOLTAGE MAG= 219.42
MIN VOLTAGE MAG= 197.43
MAX PHASE SHIFT= 3.44
MAX CURRENT= 211.66
WATTS LOST= 4547
CABLE MASS(lbm)=1136.84
VOLT SOURCE,p= 220 .00
FREQUENCY= 20000

CABLE PARAMETERS

MAIN FEEDERS		TIE FEEDERS	
r1=	1.5E-03	r2=	1.9E-03
L1=	1.8E-08	L2=	1.8E-08
c1=	3.0E-09	c2=	3.0E-09
amp=	90	amp=	73
kg/m=	0.588	kg/m=	0.465

nasa cable

LOAD DATA					LOCATION	KW	P.F.	MAX
L1	98.84	-55.41	113.31	-0.51	LKS	25.00	0.90	25.00
L2	51.57	-26.37	57.92	-0.47	LBS	11.90	0.90	16.90
L3	49.57	-26.36	56.15	-0.49	LBP	11.90	0.90	16.90
L4	55.60	-27.55	62.06	-0.46	LKP	12.50	0.90	17.50
L5	44.59	-21.73	49.60	-0.45	TBP	9.90	0.90	14.90
L6	75.96	-38.27	85.05	-0.47	UKP	17.30	0.90	22.30
L7	62.30	-31.70	69.90	-0.47	UBP	14.30	0.90	19.30
L8	59.83	-31.96	67.83	-0.49	UBS	14.30	0.90	19.30
L9	36.32	-19.14	41.50	-0.48	UKS	8.60	0.90	13.60
L10	40.38	-22.19	46.25	-0.50	TBS	9.90	0.90	14.90

source 580.87 0.57 580.87 .00 IMPEDANCE,r= 0.001 .00

LOAD ANALYSIS (RADIAL, 208 VAC)

TABLE 7.20-10

Taking switchgear quantity into consideration, however, the RING and RADIAL configurations result in lower RBI quantities. Table 7.20-11 summarizes switchgear necessary for each architecture. The switchgear quantities are based on the RBI assumptions indicated in Figure 7.20-6.

TABLE 7.20-11. SWITCHGEAR SUMMARY

	RING	NETWORK	STAR	RADIAL
Switchgear Quantity	34	40	50	34

The final criteria, fault protection methods, is not as easy to quantify as the previous criteria discussed. Table 7.20-12 along with Figures 7.20-7, -8, -9, and -10 indicate the various fault protection methods available to detect and isolate a hard fault. In all cases, these fault protection methods are in current use today on all utility grids. The application of these methods to the Space Station is straightforward and each configuration will use a combination of the various methods. Therefore, the method of fault protection is not a significant factor in the distribution architecture selection.

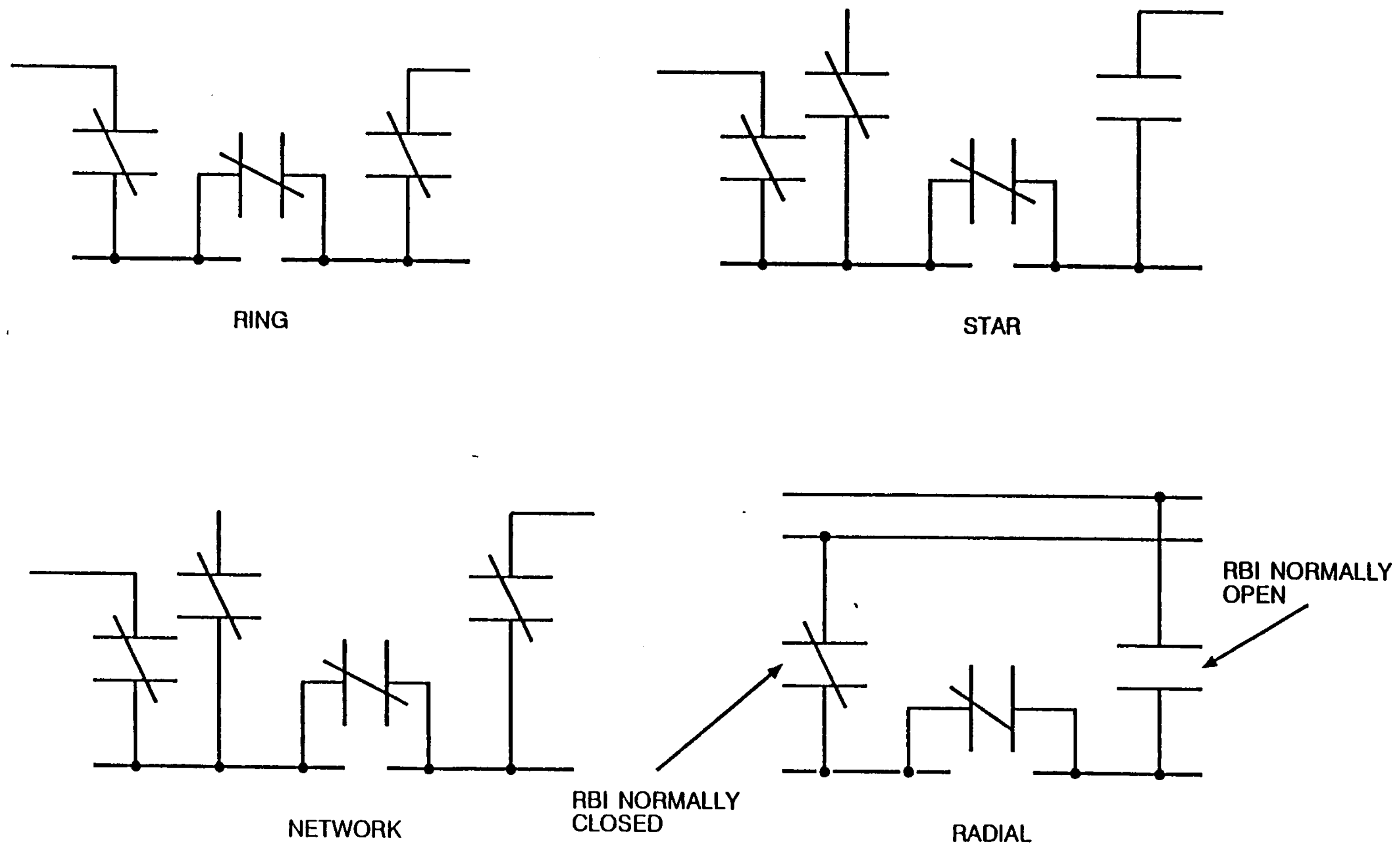
TABLE 7.20-12. FAULT PROTECTION METHODS SUMMARY

	RING	NETWORK	STAR	RADIAL
Differential*	X (380')	X (310')	X	X (480')
Impedance	X	X	X	
Reverse Power				
Overcurrent			X	X

*( ) indicates length of control wires needed.

An additional issue for consideration that has been brought up intermittently, is current imbalance between the "hot" wire and the "return" wire in a single phase system. This is an important issue because this current imbalance can result in the generation of EMI by the power system and possibly exceeding Space Station requirements. This issue, however, is a common problem to all the architectures being evaluated due to the fact that redundant feeders are required to each load bus in order to meet redundancy and failure criteria. Therefore, it is not a factor in the distribution architecture trade

RBI ASSUMPTIONS



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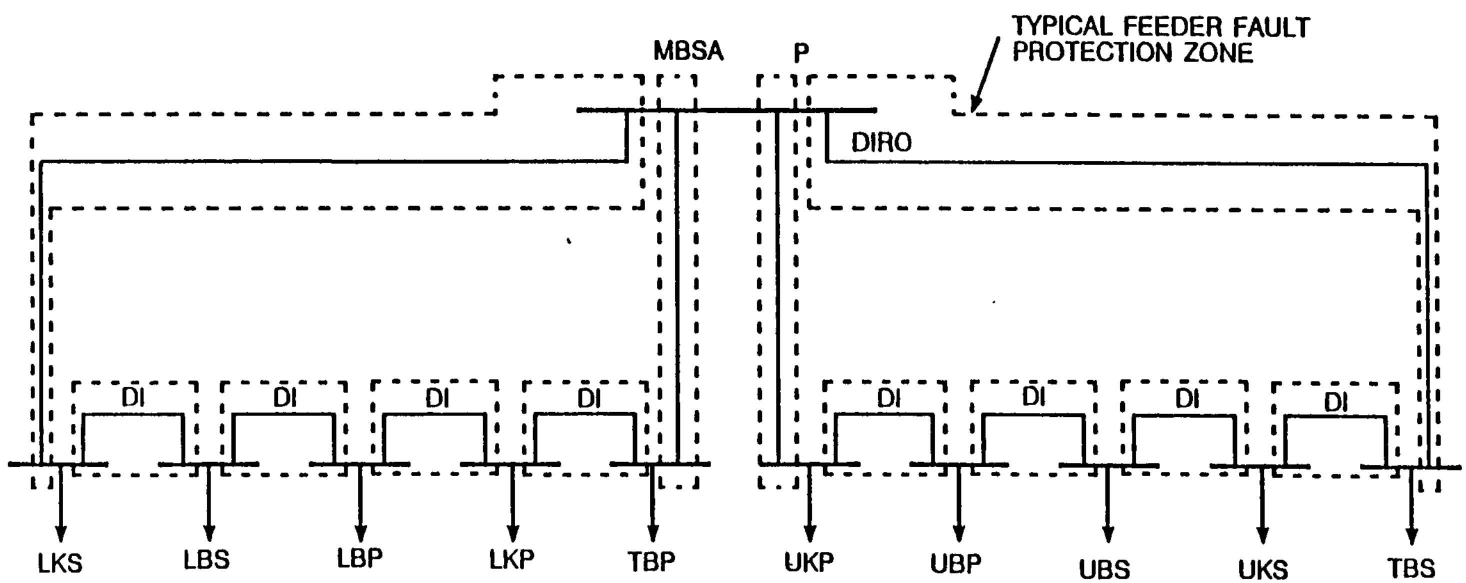
Rocketdyne Division

Figure 7.20-6

RING



SYSTEM VOLTAGE

	450	220
MASS (lbm)	390	797
LOSSES (watts)	1980	4020
LINE V RANGE	450-437	219-207
RBI'S	34	34

PROTECTION
D = DIFFERENTIAL I = IMPEDANCE R = REVERSE POWER O = OVERCURRENT

Q86D-13-1475

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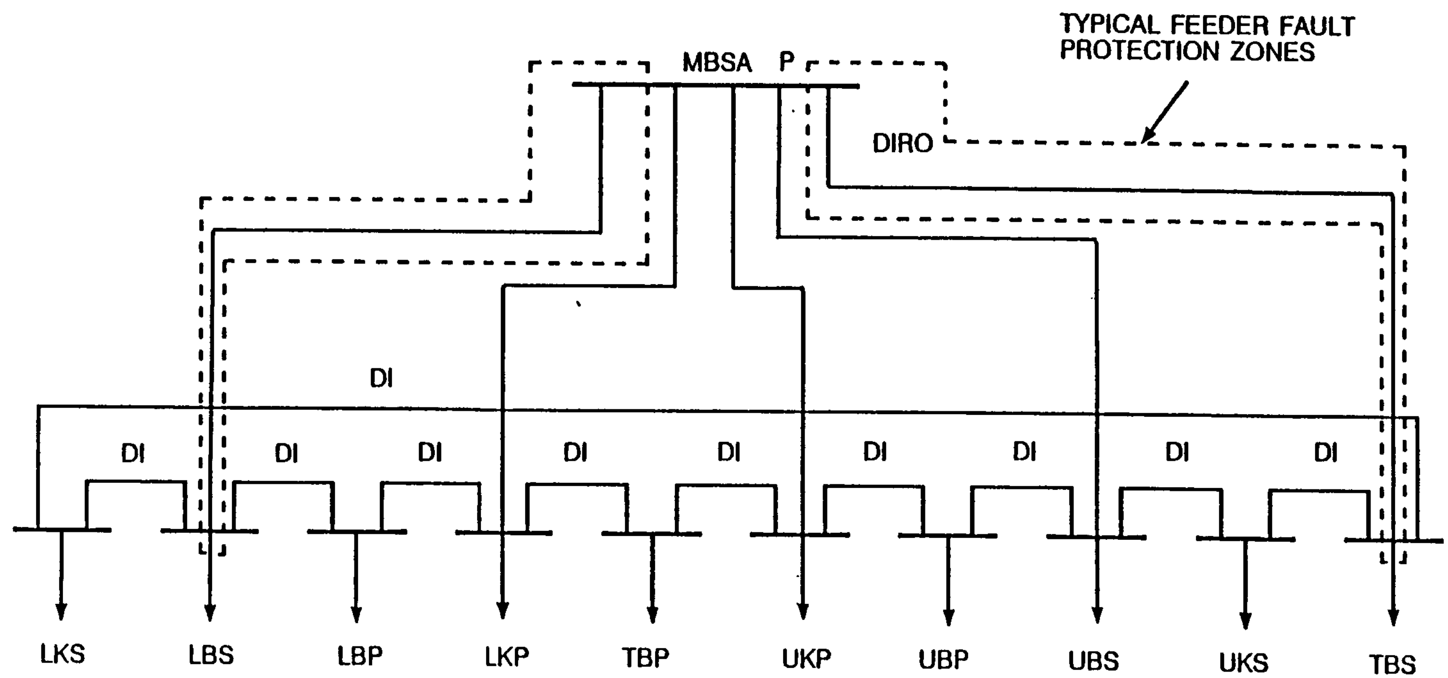
Rocketdyne Division

Figure 7.20-7

NETWORK



SYSTEM VOLTAGE

	450	220
MASS (lbm)	404	826
LOSSES (watts)	2100	4298
LINE V RANGE	450-438	219-207
RBI'S	40	

PROTECTION
D = DIFFERENTIAL
I = IMPEDANCE
R = REVERSE POWER
O = OVERCURRENT

Q86D-13-1474

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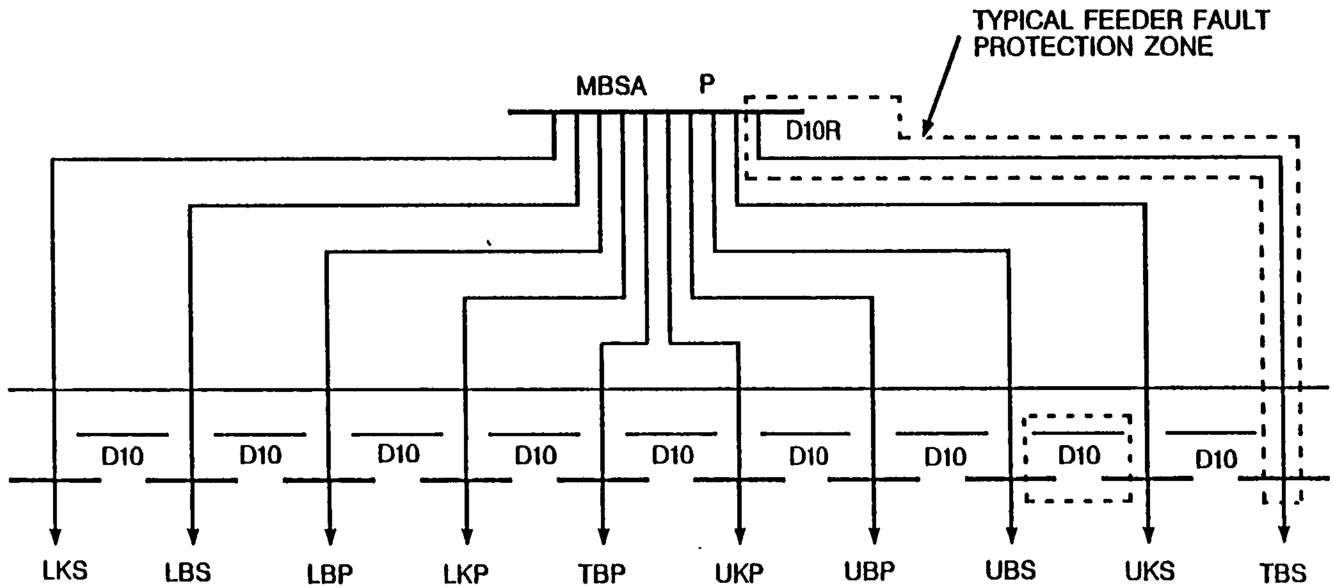
Rocketdyne Division

Figure 7.20-8

STAR



SYSTEM VOLTAGE

	450	220
MASS (lbm)	507	1038
LOSSES (watts)	1728	3371
LINE V RANGE	450-436	219-206
RBI'S	50	50

PROTECTION
D = DIFFERENTIAL I = IMPEDANCE R = REVERSE POWER O = OVERCURRENT

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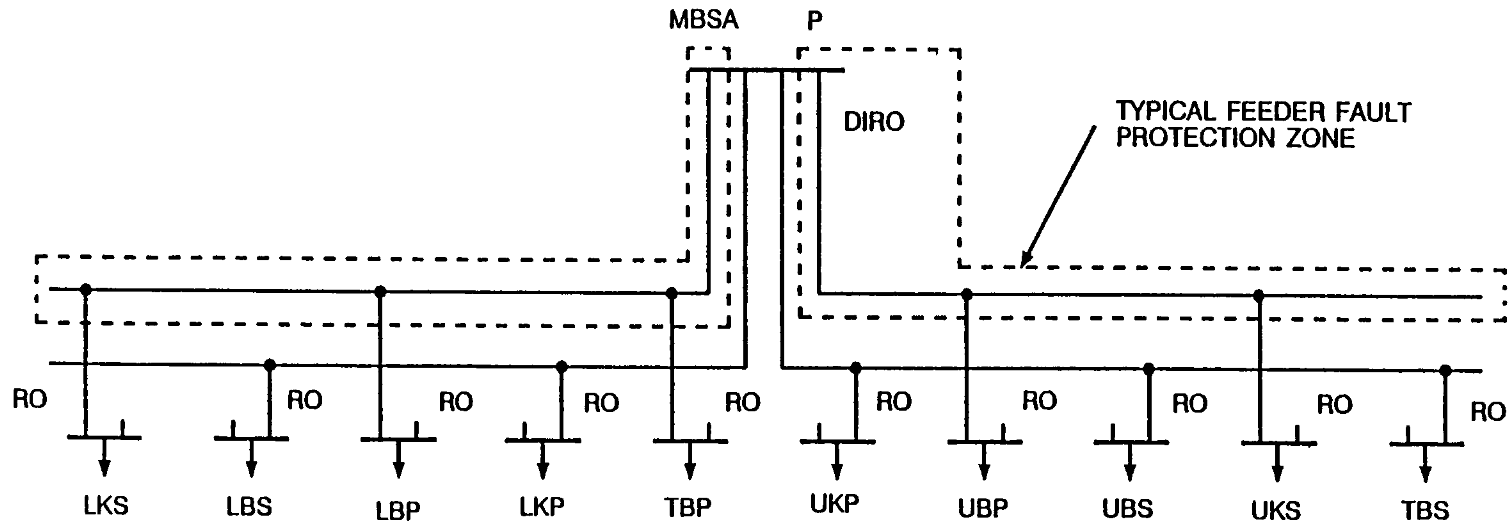
Rocketdyne Division

Figure 7.20-9

RADIAL



SYSTEM VOLTAGE

	450	220
MASS (lbm)	556	1137
LOSSES (watts)	2294	4547
LINE V RANGE	450-428	219-197
RBI'S	34	34

PROTECTION
D = DIFFERENTIAL I = IMPEDANCE R = REVERSE POWER O = OVERCURRENT

Q86D-13-1472

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Rocketdyne Division

Figure 7.20-10

study, but rather an input into the design requirements for the RBIs and for system operation. Additional work in the test beds is needed in this area to quantify the current imbalance levels necessary to create significant EMI problems and to test the EMI effects created by the cable design.

7.20.6 Conclusions

Based on criteria of mass, efficiency, protection methods, and switchgear requirements, the RING architecture is recommended for the baseline Space Station PMAD. This architecture has the least mass, good efficiency, low switchgear count, and also has the advantage of being flexible in that it can be configured to operate as a RING or a RADIAL depending upon switchgear positions. The STAR configuration is also attractive since it represents the most efficient architecture. It can be operated as a RING or a NETWORK making it the most flexible of all architectures. The penalty, however, is added weight and additional RBIs. The RADIAL configuration is the least desirable of all architectures since it is the heaviest, least efficient, and least flexible.

7.21 PMAD Load Analysis

In order to properly size the distribution equipment capacity for the external (non-manned module) station areas, knowledge of the expected loads is required. This information is now starting to be generated by the other centers working on payloads and housekeeping (subsystem) loads. Based on information from C&A Panel Meetings for utility ports, a preliminary loads analysis for the electric power system has begun.

Figure 7.21-1 indicates PDCA locations and zones served along with the location and connected load values for loads expected on the external station. Table 7.21-1 lists the abbreviations, quantities, and power levels for loads. Table 7.21-2a through j tabulates the maximum demand load that could be expected at each PDCA during station life.

After reviewing Table 7.21-2a-j for PDCA loading, it becomes apparent that a PDCA capacity of 50 kW (or 25 kW/PDCU) is needed to serve the external station loads. All of the PDCA's would not be loaded to their maximum value at the same time since this would exceed the generation capacity of the station, however, it is possible and probable that one PDCA would be delivering its maximum load while the others were very lightly loaded. In order for the space station EPS to serve as a utility, the PDCA's must be designed to meet these changing conditions and sized for their maximum expected loading.

Carrying the load calculation further, Table 7.21-3a and b tabulates the maximum demand loading that could be expected on the lower ring feeder network and the upper ring feeder network. The values indicate that 50 kW feeders will be necessary to deliver the required power under maximum demand loading conditions This will also require that the PDCU power buses that are in series with the feeders be sized at 50 kW, which is greater than that needed to serve the loads connected to the PDCU (previously stated as 25 kW).

Note that the values calculated are for the growth station. Maximum demand loading of the rings would be significantly reduced at IOC since only 5 payload attach points would be operational. However, the maximum demand load at each PDCA would be close to the values shown in Table 7.21-2a - j.

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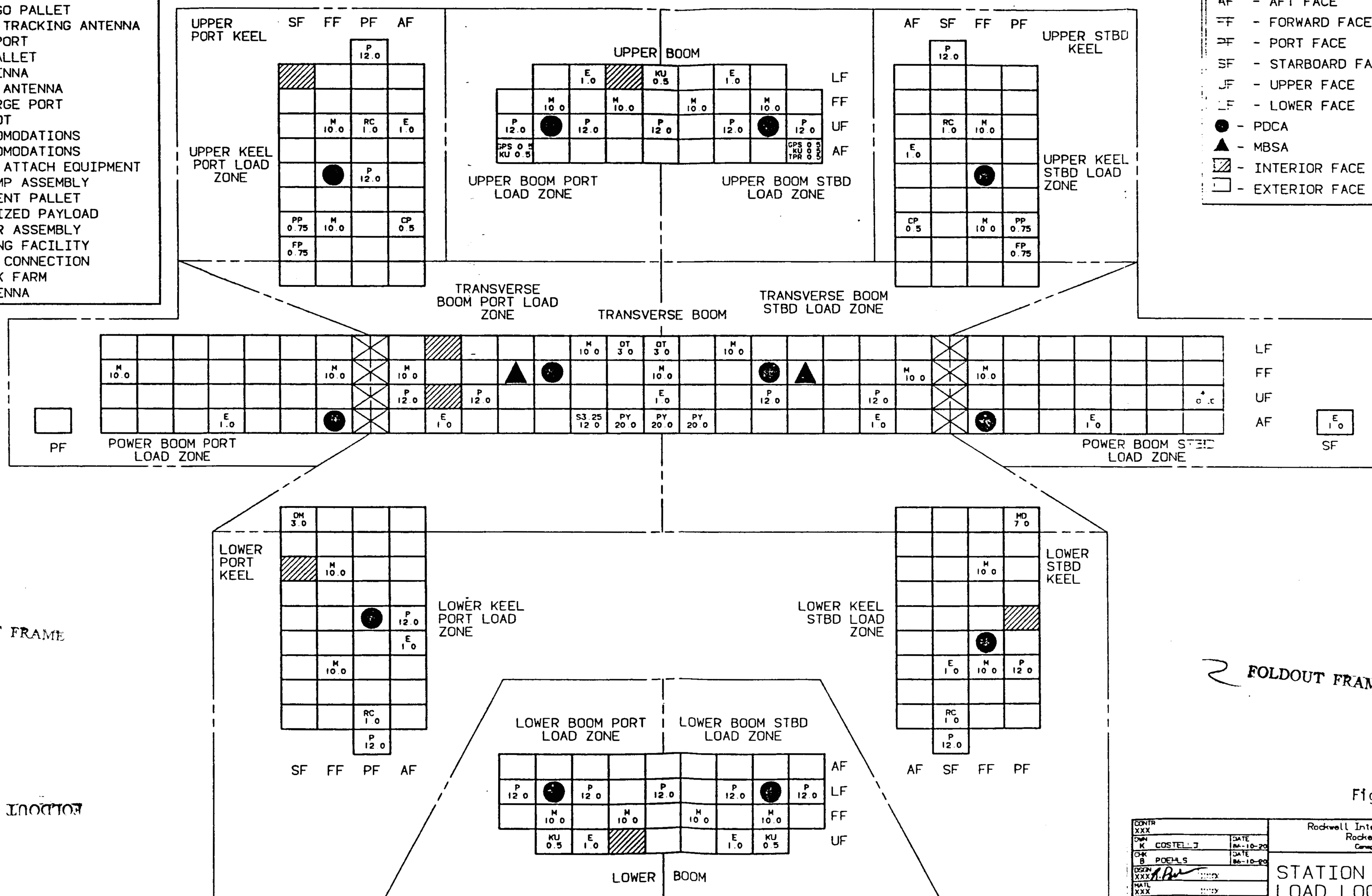
STATION LOAD LOCATIONS

LEGEND (LOADS)	
ACA	ATTITUDE CONTROL ASSEMBLY
CP	DRY CARGO PALLET
C&T	COM AND TRACKING ANTENNA
E	EVA SUPPORT
FP	FLUID PALLET
GPS	GPS ANTENNA
KU	KU BAND ANTENNA
M	MSC CHARGE PORT
MD	MSC DEPOT
OM	OMV ACCOMODATIONS
OT	OTV ACCOMODATIONS
P	PAYLOAD ATTACH EQUIPMENT
PA	ATCS PUMP ASSEMBLY
PP	PROPELLENT PALLET
PY	PRESSURIZED PAYLOAD
RC	THRUSTER ASSEMBLY
S	SERVICING FACILITY
SS	ORBITER CONNECTION
TF	OTV TANK FARM
TDR	TDR ANTENNA

LEGEND (STRUCTURE)	
AF	AFT FACE
FF	FORWARD FACE
PF	PORT FACE
SF	STARBOARD FACE
UF	UPPER FACE
LF	LOWER FACE
●	PDCA
▲	MBSA
(hatched box)	INTERIOR FACE
(open box)	EXTERIOR FACE



FOLDOUT FRAME

FOLDOUT FRAME

Figure 7.21-1

CONTR XXX	Rockwell International Corporation, Rocketdyne Division, Canoga Park, California
DWN K. COSTELLI DATE 86-10-20	
CHK B. POEHLS DATE 86-10-20	STATION LOAD LOCATIONS
DSGN XXX	
MATL XXX	
STRUCT XXX	
DESIGN ACTIVITY APVD XXX	SIZE D FSCM NO 02602 DWG NO 7R070019

To summarize, the loads analysis indicates that the ring feeder capacity should be 50 kW for each feeder cable and the PDCU bus size should also be 50 kW. Options to the above would be to revise the system architecture to a star system or a tapped radial. For the star, this would permit the use of 25 kW cables and PDCU's. For the tapped radial, 50 kW feeders would be needed but the 25 kW PDCU could be used.

TABLE 7.21-1

LOADS	ABBREV	QTY IOC	QTY GROWTH	PWR (KW) IOC	PWR (KW) GROWTH	NOTES
PAYLOAD ATTACH EQUIPMENT	P	20	20	12	12	a
PRESSURIZED PAYLOADS	PY	1	3	20	30	
MSC CHARGING POINTS	M	28	30	10	10	b
MSC MAINTENANCE DEPOT	MD	1	1	7	7	
SERVICING FACILITY	S	2	2	3.25/12	3.25/12	c
OMV ACCOMMODATIONS	OM	1	2	3	3.5	
OTV ACCOMMODATIONS	OT	0	2	0	3	
OTV TANK FARM	TF	0	1	0	5	
LOGISTICS PALLETS						
DRY CARGO PALLET	CP	2	2	.5	.5	
FLUIDS PALLET	FP	2	2	.75	.75	
PROPELLANTS PALLET	PP	2	2	.75	.75	
EVA SUPPORT (TOOL CHARGING)	E	13	15	1	1	d
ORBITER	SS	2	2	10	10	

a) At IOC only 5 payload attach equipment points will be provided, but capacity to grow to 20 will be provided.

b) only one MSC charging point will be used at any one time. The 12 kw includes 1.8kw for payload support. 2.9 kW average for battery charge (12 hours),5.3 kW peak (minutes) for MSC 2-3 kW average).

c) The split power Panel indicates 3.25kw for the servicing facility and 12 kw to power the payload being serviced.

d) Only one tool charging port will be used at any one time.

TABLE 7.21-2a

PDCA LOAD SCHEDULE

LOCATION - LBS (lower Boom starboard)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	2.5	12	1	30
MSC Charging Points	M	2	10	0.5	10
EVA Support	E	1	1	1	1
Ku Antenna	Kn	1	0.5*	1	0.5
				TOTAL	41.5 kW

* Estimate

TABLE 7.21-2b

PDCA LOAD SCHEDULE

LOCATION - LBP (lower boom port)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	2.5	12	1	30
MSC Charging Points	M	2	10	0.5	10
EVA Support	E	1	1	1	1
Ku Antenna	Kn	1	0.5*	1	0.5
				TOTAL	41.5 KW

* Estimate

TABLE 7.21-2c

PDCA LOAD SCHEDULE

LOCATION - LKS (lower keel starboard)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	2	12	1	24
MSC Charging Points	M	2	10	0.5	10
EVA Support	E	1	1	1	1
Thruster Assembly	RC	1	1*	1	1.0
				TOTAL	36 kW

* Estimate

TABLE 7.21-2d

PDCA LOAD SCHEDULE

LOCATION - LKP (lower keel port)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	2	12	1	24
MSC Charging Points	M	2	10	0.5	10
EVA Support	E	1	1	1	1
Thruster Assembly	RC	1	1.0A	1	1.0
				TOTAL	36 kW

* Estimate

TABLE 7.21-2e

PDCA LOAD SCHEDULE

LOCATION - TBS (transverse boom starboard)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	2	12	1	24
MSC Charging Points	M	2.5	10	0.4	10
ATCS Pump Assy	PA	1	3	1	3
MSC Depot	MD	1	7	1	7
OTV Accommodations	OT	1	3	1	3
EVA Support	E	1	1	1	1
				TOTAL	48 kW

* Estimate

TABLE 7.21-2f

PDCA LOAD SCHEDULE

LOCATION - TBP (transverse boom port)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	1	12	1	12
MSC Charging Points	M	2.5	10	0.4	10
OTV Accommodations	OT	1	3	1	3
EVA Support	E	2	1	0.5	1
ATCS Pump Assy	PA	1	3*	1	3
Altitude Control Assy	ACA	1	4*	1	4
OMV Accommodations	OM	1	3	1	3
Satellite Servicing	S	1	15.25	1	15.25
				TOTAL	51.25 kW

* Estimate

TABLE 7.21-2g

PDCA LOAD SCHEDULE

LOCATION - UKS (upper keel starboard)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	1	12	1	12
MSC Charging Points	M	2	10	.5	10
Thruster Assy	RC	1	1*	1	1
EVA Support	E	1	1	1	1
Propellant Pallet	PP	1	0.75	1	0.75
Fluid Pallet	FP	1	0.75	1	0.75
Cargo Pallet	CP	1	0.5	1	0.5
				TOTAL	26 kW

* Estimate

TABLE 7.21-2h

PDCA LOAD SCHEDULE

LOCATION - UKP (upper keel port)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	2	12	1	24
MSC Charging Points	M	2	10	.5	10
Thruster Assy	RC	1	1*	1	1
EVA Support	E	1	1	1	1
Propellant Pallet	PP	1	0.75	1	0.75
Fluid Pallet	FP	1	0.75	1	0.75
Cargo Pallet	CP	1	0.5	1	0.5
				TOTAL	38 kW

* Estimate

TABLE 7.21-2i

PDCA LOAD SCHEDULE

LOCATION - UBS (Upper boom starboard)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	2.5	12	1	30
MSC Charging Points	M	2	10	.5	10
EVA Support	E	1	1	1	1
GPS Antenna	GPS	1	0.5*	1	0.5
Ku Antenna	Ku	1	0.5*	1	0.5
TDR Antenna	TDR	1	0.5*	1	0.5
				TOTAL	42.5 kW

* Estimate

TABLE 7.21-2j

PDCA LOAD SCHEDULE

LOCATION - UBP (upper boom port)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	2.5	12	1	30
MSC Charging Points	M	2.5	10	.4	10
EVA Support	E	1	1	1	1
GPS Antenna	GPS	1	0.5*	1	0.5
Ku Antenna	Ku	2	0.5*	1	1.0
				TOTAL	42.5 kW

* Estimate

TABLE 7.21-3a
MAX LOADING LOWER RING
(LBP, LBS, LKP, LKS, TBS)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	11	12	1	121
MSC Charging Points	M	10.5	10	.09	10
EVA Support	E	5	1	.2	1
Thruster Assembly	RC	2	1*	1	2
Ku Antenna	Ku	2	0.5*	1	1
ATCS Pump Assembly	PA	1	3*	1	3
MSC Depot	MD	1	7	1	7
				TOTAL	145 kW

* Estimate

TABLE 7.21-3b
MAX LOADING UPPER RING
(UBP, UBS, UKP, UKS, TBP)

LOAD	ABB	QTY	PWR (KW)	MAX DEMAND FACTOR	MAX DEMAND (KW)
Payload Attach Equipment	P	9	12	1	108
MSC Charging Points	M	11	10	.09	10
OTV Accommodations	OT	1	3	1	3
EVA Support	E	6	1	.17	1
ATCS Pump Assembly	PA	1	3*	1	3
Altitude Control Assembly	ACA	1	4*	1	4
OMV Accommodations	OM	1	3	1	3
Thruster Assembly	RC	2	1*	1	2
Servicing Facility	S	2	15.25	1	30.5
Propellant Pallet	PP	2	0.75	1	1.5
Fluid Pallet	FP	2	0.75	1	1.5
Cargo Pallet	CP	2	0.50	1	1.0
GPS Antenna	GPS	2	0.5*	1	1.0
Ku Antenna	Ku	3	0.5*	1	1.5
TDR Antenna	TDR	1	0.5*	1	0.5
				TOTAL	171.5 kW

* Estimate

8.0 COST DRIVERS

In order to determine EPS cost drivers for the previous submittal of DR02 (June 30, 1986), the Rocketdyne life-cycle cost (LCC) model was run to identify the more significant costs and the factors contributing to them. At that time, the model was run using the following assumptions and ground rules.

One station plus one platform
Station power: 75 kW IOC, 300 kW growth
Platform power: 8 kW IOC, 15 kW growth
IOC station has 2-12.5 kW PV modules and 2-25 kW SD modules
Station growth is by replication of SD modules
Station and platform commonality for PV arrays and Ni-H_2 batteries
Beta joints are included
PMAD frequency: 20 kHz station, 20 kHz platform
User load converters are included.
All costs include estimates for subcontractor and contractor G&A and fee and other WBS items (management, SE&I, GSE, IACO, Test, Ops, Maint., etc.)

For this submittal, the primary changes made were:

(1) Station reboost cost was omitted.

(2) Costs were estimated for the latest PMAD architecture (including 20 kHz equipment on both the station and platform) and include user load converters.

The reason for omitting station reboost cost is that the station propulsion system now uses hydrogen-oxygen fuel, and the fuel source for reboost is already on the station (i.e., water). The only reboost cost is for (1) the extra electrolysis units required to electrolyze water into hydrogen and oxygen and (2) the power to operate the units. An estimate showed these costs to be small, and they have not been included at this time since they do not significantly affect the cost drivers. The only reboost cost included is for the platform.

8.1 RESULTS

A detailed breakdown of cost distribution is provided in Table 8-1. It shows costs as a percentage of total LCC. The primary cost drivers can be obtained from this table.

The largest cost driver is replacement hardware cost during 30-years of operations (36% of the total LCC for station + platform). The factors in this cost are:

Quantity of each ORU (orbital replacement unit)
Mean time between replacement (MTBR) for each ORU
Cost of each ORU (for hardware cost)
Weight of each ORU (for launch cost)

For the station, launch costs are about 40% of the total replacement hardware cost and are dependent on the orbital altitude.

Shown below are: (1) the ORUs that are the primary contributors to replacement hardware costs, (2) the contribution to the subsystem life-cycle cost (LCC), (3) the important cost factors.

SUBSYSTEM	ORU	% CONTRIBUTION TO SUBSYSTEM LCC	PRIMARY FACTORS THAT CONTRIBUTE TO COST
PV MODULES	SOLAR ARRAY WING	52%	ORU COST
	Ni-H2 Battery	20%	ORU QTY,WEIGHT,MTBR
		72%	
SD MODULES	CONCENTRATOR SURFACE	26%	ORU COST & WEIGHT
	RECEIVERS/PCU	14%	ORU COST & WEIGHT
	RADIATOR PANEL	39%	ORU QTY & WEIGHT
		79%	
PMAD	POWER DISTRIBUTION & CONTROL UNIT (PDCU)	89%	ORU QUANTITY

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TABLE 8.1 EPS COST DRIVERS -- % OF TOTAL LCC

	STATION								1 PLATFORM							
	IOC POWER(75KW)				ADDED POWER(225KW)				IOC POWER(8KW)				ADDED POWER(8KW)			
IOC COST				32.94				18.49				2.04				1.33
DDT&E			20.10				0.00				0.09				0.00	
PV MODULE		5.40				0.00				0.05				0.00		
PV WING ARRAY	1.27				0.00				0.02				0.00			
NI-H2 BATTERY	0.77				0.00				0.02				0.00			
THERMAL CONTROL	0.49				0.00				0.00				0.00			
PV PMAD	2.36				0.00				0.00				0.00			
BETA JOINT	0.52				0.00				0.00				0.00			
SD MODULE		8.38				0.00				0.00				0.00		
CONCENTRATOR	2.40				0.00				0.00				0.00			
RECEIVER/PCU	2.67				0.00				0.00				0.00			
RADIATOR	1.51				0.00				0.00				0.00			
SD PMAD	1.80				0.00				0.00				0.00			
PMAD		4.90				0.00				0.04				0.00		
SOFTWARE		1.42				0.00				0.00				0.00		
PRODUCTION			7.41				10.55				1.45				1.08	
PV MODULE		2.45				0.00				1.08				1.08		
PV WING ARRAY	1.27				0.00				0.64				0.64			
NI-H2 BATTERY	0.49				0.00				0.25				0.25			
THERMAL CONTROL	0.11				0.00				0.00				0.00			
PV PMAD	0.50				0.00				0.16				0.16			
BETA JOINT	0.08				0.00				0.04				0.04			
SD MODULE		1.60				8.73				0.00				0.00		
CONCENTRATOR	0.40				2.86				0.00				0.00			
RECEIVER/PCU	0.17				1.22				0.00				0.00			
RADIATOR	0.76				3.42				0.00				0.00			
SD PMAD	0.23				1.05				0.00				0.00			
BETA JOINT	0.04				0.18				0.00				0.00			
PMAD		3.35				1.82				0.38				0.00		
OTHER			5.44				7.94				0.49				0.25	
INITIAL SPARES		1.92				0.00				0.15				0.00		
PV MODULE	0.96				0.00				0.00				0.00			
SD MODULE	0.82				0.00				0.00				0.00			
PMAD	0.14				0.00				0.15				0.00			
LAUNCH		2.86				6.41				0.32				0.25		
PV MODULE	0.86				0.00				0.25				0.25			
SD MODULE	1.30				5.83				0.00				0.00			
PMAD	0.69				0.57				0.07				0.00			
DEPLOYMENT		0.07				0.26				0.02				0.00		
PV MODULE	0.01				0.00				0.02				0.00			
SD MODULE	0.06				0.26				0.00				0.00			
SYSTEM IMPACTS		0.59				1.28				0.00				0.00		
PV MODULE	0.31				0.00				0.00				0.00			
SD MODULE	0.28				1.28				0.00				0.00			
30-YEAR OPERATIONS COST								37.50								7.69
REPLACEMENT HARDWARE & LAUNCH							30.34								5.74	
PV MODULE						9.55								5.37		
SD MODULE						17.06								0.00		
PMAD						3.73								0.37		
ON-ORBIT OPERATIONS & MAINT							5.28								0.38	
PV MODULE						0.54								0.28		
SD MODULE						3.28								0.00		
PMAD						1.46								0.10		
REBOOST							0.00								1.26	
PV MODULE						0.00								1.26		
SD MODULE						0.00								0.00		
PMAD						0.00								0.00		
GROUND SUPPORT							1.88								0.31	

On-orbit operations and maintenance is about 5% of the total LCC. Its primary factors are the MTBR for each ORU combined with the number of EVA and IVA hours and the cost of those hours. Any significant change to these factors could make an appreciable difference in LCC.

While DDT&E is a large part of the IOC cost, it is only 20% of the total LCC. Here is the area where relatively minor expenditures could possibly lead to major cost savings in production and 30-year operations costs, especially if ORU cost and weight can be reduced or the mean time between replacement can be increased.

For instance, an increase of 5 years in MTBR for PV arrays would reduce 30-year replacement hardware costs by about 70 million dollars. Similarly, increasing battery MTBR by 5 years would save about 180 million dollars in total LCC.

Operations costs for 2-12.5 kW PV modules is approximately 20 million dollars per year compared to 5 million dollars per year for a 25-kW SD module. The two PV modules could be replaced with one SD module for about 80 million dollars which could be recovered in operations cost savings in less than 6 years.

8.2 EVALUATION METHODOLOGY

The cost assessment logic and information flow is shown in Figure 8-1. To accomplish the cost assessment, worksheets and data tables were completed. They assure that the evaluation is based on as much factual data as practical and provide a documented record of their bases.

An electronic spreadsheet (LOTUS 1-2-3) has been used to generate and document the cost assessment data. The cost assessment spreadsheet contains several levels of worksheets and data files. Lower levels, such as the reboost cost calculations spreadsheet (see example Table 8-2), are used as inputs to upper level spreadsheets, such as the cost assessment worksheet (see example Table 8-3). The cost assessment worksheet information is, in turn, collected in the cost summary worksheet, which is used to generate the LCC.

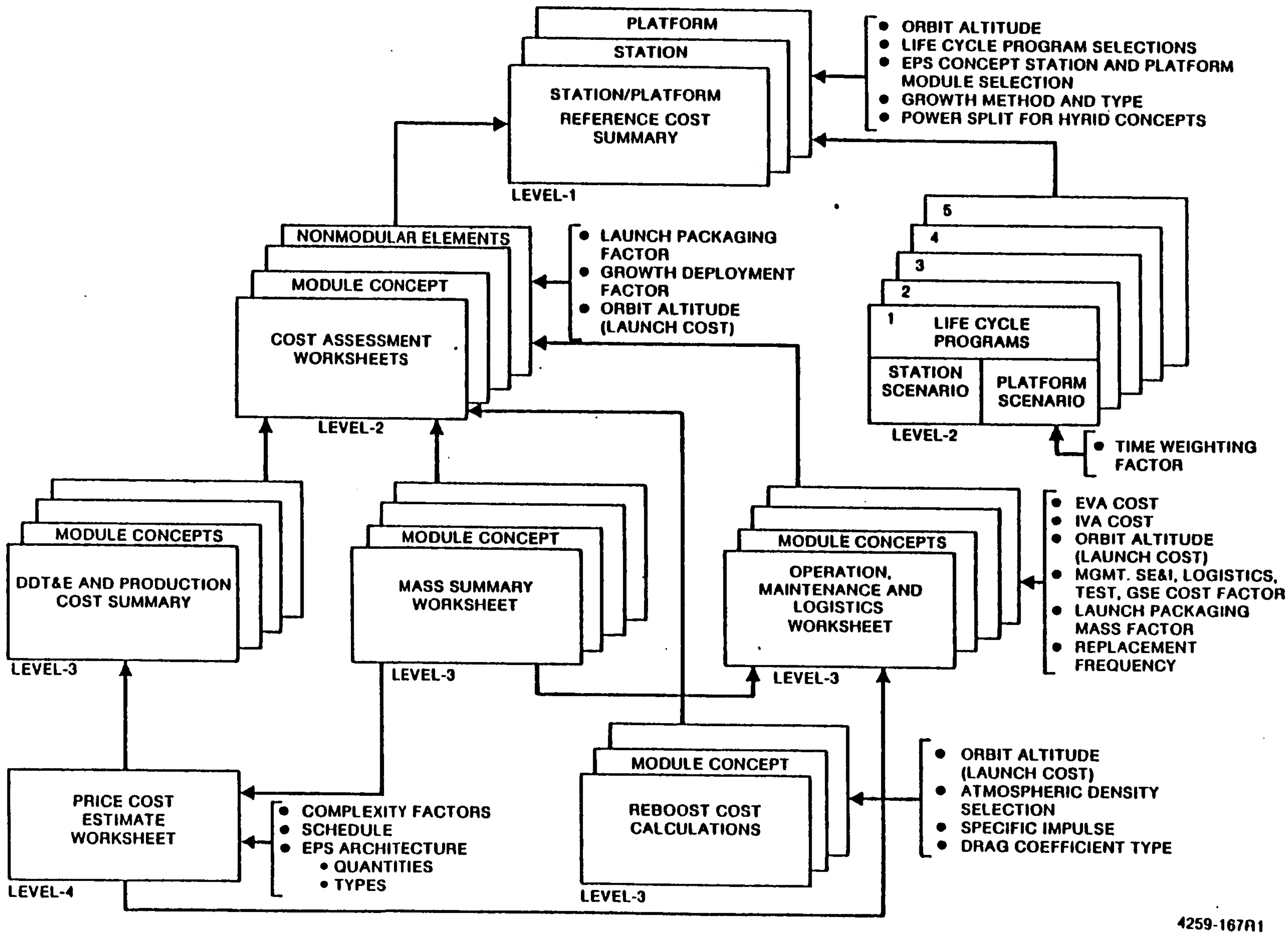

Figure 8.1. Cost Assessment Logic and Information Flow

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MODULE COMPONENT	DRAG COEF. TYPE	(b) DRAG COEF. (CD/CDN) VALUE	FUEL MASS/ AREA(WF/A) (lb mass/ ft^2	UNIT RE-BOOST COST ($/m^2/yr)	STATION PV-2H 12.5KWe 1	CBC-4 25KWe 2	ORC-4 25KWe 3	4	5	(a) PV-3 8KWe 6	7	8
PV ARRAY	4	0.5677	0.0876	2751	289.00					223.00		
SD CONCENTRATOR												
FRONTAL	4	0.5677	0.0876	2751		194.60	231.40					
PROFILE	2	0.6282	0.0969	3045		20.70	26.30					
PGS RADIATOR												
PARALLEL TO TRNSVERSE BOOM	5	0.6282	0.0969	3045								
NORMAL TO TRANSVERSE BOOM	6	0.2314	0.0357	1121		92.00	200.50					
PARASITIC LOAD RADIATOR	6	0.2314	0.0357	1121		0.67	0.67					
EPS STRUCTURE	1	1.0000	0.1543	4847	23.90	7.80	6.00			19.96		
PGS SUBTOTAL					312.90	315.77	464.87			242.96		
ESS RADIATOR(ECLIPSE AND CONTINGENCY)	2	0.6282	0.0969	3045	35.00					0.00		
ESS RADIATOR(SAVE HAVEN AND CONTINGENCY)	2	0.6282	0.0969	3045								
ESS RADIATOR(SAVE HAVEN)	2	0.6282	0.0969	3045								
ESS SUBTOTAL					35.00	0.00	0.00			0.00		
PMAD RADIATOR	2	0.6282	0.0969	3045	0.00	0.00	0.00			0.00		
TOTAL MODULE PHYSICAL AREA(SQUARE METERS)					347.90	315.77	464.87			242.96		
ANNUAL MODULE REBOOST COST($MILLION)										1.21		

EPS MODULE PHYSICAL AREA(SQUARE METERS) covers columns 1–8.

(a) TWO WING ASSEMBLIES PER MODULE
(b) DRAG CALCULATIONS DO NOT INCLUDE SHADOWING EFFECTS
(c) THE DENSITY VALUES ARE BASED ON THE 2 SIGMA,MEAN AND -2 SIGMA VALUES OF PREDICTED SOLAR FLUX

H-SPACE STATION ORBIT ALTITUDE(n.mi)	250
DENSITY SELECTION NUMBER	2
RHO-ATMOSPHERIC DENSITY(Kg/m^3)	8.00E-13
LC-LAUNCH COST($/lb mass)	2920
IPS-SPECIFIC IMPULSE(lb-s/lb mass)	230
CDN-MAXIMUM DRAG COEFFICIENT(SURFACE NORMAL TO VELOCITY VECTOR)	2.30
WF/[A(CD/CDN)]-ANNUAL FUEL MASS/ UNIT AREA-DRAG COEF PRDCT(lb mass/ft^2)	0.1543

LAUNCH COST(LC) AND ATMOSPHERIC DENSITY(RHO) MENU

H (n.mi)	LC ($/lb mass)	AVERAGED FROM 1991-2001(c) 2SIGMA (97.7%) 1	MEAN (50%) (Kg/m^3) 2	-2SIGMA (2.3%) 3	RANGE MAX. 4	RANGE MIN. 5
175	2400	2.20E-11	1.10E-11	6.00E-12	4.25E-11	3.30E-12
200	2550	1.05E-11	4.50E-12	2.15E-12	2.20E-11	1.00E-12
220	2690	5.50E-12	2.20E-12	1.11E-12	1.35E-11	4.20E-13
250	2920	2.26E-12	8.00E-13	3.00E-13	6.30E-12	1.16E-13
270	3100	1.38E-12	4.23E-13	1.38E-13	4.25E-12	5.55E-14

DRAG COEFFICIENT MENU

TYPE	BETAO (DEG)	THETA (DEG)	DESCRIPTION	CD/CDN VALUE
1	90	0	FIXED NORMAL (VELOCITY) SURFACE	1.0000
2	0	90	TRANSVERSE BOOM PARALLEL SURFACE, ALPHA JOINT FOLLOWING	0.6282
3	0	0	TRANSVERSE BOOM PERPENDICULAR SURFACE, ALPHA JOINT FOLLOWING	0.0734
4	90	0	SUN POINTING SURFACE, ALPHA AND BETA JOINT FOLLOWING	0.5677
5	0	90	NON SUN POINTING SURFACE PARALLEL TO, TRANSVERSE BOOM,ALPHA AND BETA JOINT FOLLOWING	0.6282
6	0	0	NON SUN POINTING SURFACE PERPENDICULAR TO, TRANSVERSE BOOM,ALPHA AND BETA JOINT FOLLOWING	0.2314
7	90	0	SAME AS TYPE 4 EXCEPT SURFACE PARALLEL TO THE VELOCITY VECTOR DURING ECLIPSE(e)	0.4285
8	0	0	SAME AS TYPE 6 EXCEPT SURFACE NORMAL TO THE VELOCITY VECTOR DURING ECLIPSE(e)	0.2130
9	0	90	SAME AS TYPE 5 EXCEPT SURFACE NORMAL TO THE VELOCITY VECTOR DURING ECLIPSE(e)	0.8100
[illegible]	0	90	SAME AS TYPE 2 EXCEPT SURFACE NORMAL TO THE VELOCITY VECTOR DURING ECLIPSE(e)	0.8100

Table 8-2. Example of Reboost Cost Calculations Spreadsheet (values not current)

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Table 8-3. Example of Cost Assessment Worksheet

COST ASSESSMENT WORKSHEET									
ORBIT ALTITUDE: 250 n.mi DS: 2		MODULE COSTS(1987 $MILLIONS)							
	MODULE NO.	STATION 1	2	3	STATION 4	STATION 5	PLATFORM 6	PLATFORM 7	8
	TYPE	PV	CBC	ORC	COM PMAD	COM PMAD	PV	COM. PMAD	NO GROWTH
	POWER(KWe)	12.50 (a)	25.00	25.00	IOC	250.00 DELTA GROW	8.00	NO GROWTH	
IOC-DDT&E FIXED									
PV MODULE(W/PV SOURCE PMAD)									
SD MODULE(W/SD SOURCE PMAD)									
COMMON PMAD									
SOFTWARE									
PSF HARDWARE									
BETA JOINT									
ALPHA JOINT									
PMAD USER LOAD CONVERTERS									
SUBTOTAL DDT&E FIXED									
IOC-PRODUCTION FIXED									
ALPHA JOINT									
PROTOFLIGHT CREDIT									
SUBTOTAL PRODUCTION FIXED									
IOC-OTHER FIXED(INITIAL SPARES)									
TOTAL FIXED									
IOC-PRODUCTION VARIABLE									
PV MODULE(W/PV SOURCE PMAD)									
SD MODULE(W/SD SOURCE PMAD)									
COMMON PMAD									
BETA JOINT									
PMAD USER LOAD CONVERTERS									
SUBTOTAL PRODUCTION VARIABLE									
IOC-OTHER VARIABLE									
SYSTEM IMPACTS									
LAUNCH LAUNCH PKG FACTOR:	1.20								
DEPLOYMENT									
SUBTOTAL OTHER VARIABLE									
TOTAL VARIABLE									
GROWTH-DELTA PRODUCTION VARIABLE($M/KW)									
GROWTH-DELTA TOTAL VARIABLE($M/KW)									
ANNUAL-FIXED(GROUND SUPPORT)									
ANNUAL-VARIABLE									
REPLACEMENT HARDWARE & LAUNCH									
ON-ORBIT OPERATIONS & MAINTENANCE									
REBOOST & OTHER SYSTEM IMPACTS(b)									
SUBTOTAL ANNUAL VARIABLE									
GROWTH-DELTA ANNUAL VARIABLE($M/KW/YR)									

(a) TWO WING ASSEMBLIES PER MODULE
(b) DRAG CALCUL[illegible]S DO NOT INCLUDE SHADOWING EFFECTS

8.2.1 Cost Assessment Worksheet

The cost assessment worksheet (see example Table 8-3) and EPS life-cycle program data are used to calculate the EPS concept cost elements appearing in the cost summary. There are three groups of cost data listed for each defined concept module, initial cost, growth cost, and annual costs. The PV and SD cost elements are multiplied by the number of modules to obtain the concept costs. Cost items such as DDT&E, initial spares, and ground support are not a function of the number of concept modules. There are other cost elements that do not grow as a function of number of modules, such as portions of PMAD. The cost assessment worksheet "pulls up data" from the operations, maintenance, and logistics (OML) worksheet, mass summary worksheet, and reboost cost calculations worksheet. Presently, the DDT&E, production, and initial spares cost data are directly inputted based on subcontractor data and PRICE-generated data.

8.2.2 Reboost Cost Calculations

As discussed in Section 8.0, the reboost cost was calculated only for the platform since the station fuel is now hydrogen-oxygen and station reboost cost is not significant. The factors that affect the reboost cost are physical surface area, surface orientation, reboost fuel specific impulse, and orbit altitude. The reboost cost is directly proportional to atmospheric density and drag coefficient and inversely proportional to reboost fuel specific impulse. The surface orientation and location determine the drag coefficient. For the same physical area, orientation and location can have a large impact on reboost cost.

The fuel launch cost and atmospheric density are functions of the orbit altitude. A predicted mean value (over a 10-year period) of predicted atmospheric density was used in the cost assessment. The uncertainty in predicting the solar flux and geomagnetic index, which affect atmospheric density, could result in a large uncertainty in reboost cost. Selection of the orbit altitude, drag type, and density selection number automatically enters the appropriate values from the menus into the spreadsheet calculations. The reboost fuel specific impulse, module, physical areas, and maximum drag coefficient are entered directly.

8.2.3 Operation, Maintenance, Logistics (OML) Worksheets

An OML worksheet is prepared for each EPS concept module. The OML worksheet calculates the ORU replacement hardware cost and its launch and maintenance costs for each ORU and totals them for the module based on the following inputs: (1) number of ORUs per module, (2) ORU unit cost, (3) ORU mass, (4) MTBR, (5) EVA and IVA maintenance times. The EVA and IVA times for module deployment are estimated providing EPS deployment cost data. The calculated data from the OML worksheet is collected in the OML worksheet summary (see example Table 8-4).

8.2.4 Mass Summary Worksheet

The mass summary worksheet lists masses by subsystem, assembly, and component (if available) for each concept module and nonmodular element. These data are used to calculate the EPS mass for input to the cost assessment worksheet (to calculate launch cost), to the OML worksheets, and to PRICE. Table 8-5 is an example of the worksheet that summarizes the mass data.

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COST ASSESSMENT STUDY INDEPENDENT INPUT PARAMETERS				COST ASSESSMENT STUDY DEFENDENT INFUT FARAMETERS	
OML WORKSHEET(OML SPREADSHEET)		COST ASSESSMENT WORKSHEET		LCD-LAUNCH COST DEPLOYMENT($/lb mass):	2690
LAUNCH PACKAGING MASS FACTOR:	1.20	LAUNCH PACKAGING MASS FACTOR:	1.20	LC-LAUNCH COST($/lb mass):	2920
EVA COST($K/man-h):	103.00	GROWTH DEPLOYMENT FACTOR:	1.00	RHO-ATMOSPHERIC DENSITY(Kg/m^3):	8.00E-13
IVA COST($K/man-h):	15.20	DEPLOYMENT ALITUTUDE(n.mi):	220	WF/A*CD-ANNUAL FUEL MASS/AREA-DRAG COEF(lb mass/ft^2):	0.1543
SUBCONTRACTOR AND RK G&A FACTOR:	1.50	PSF HARDWARE INCLUDED(Y=1,N=0):	0	NUMBER OF PLATFORMS	1
MGMT.,SE&I,LGSTCS.,TEST & GSE FACTOR:	1.50	USER LOAD CONVERTERS INCLUDED(Y=1,N=0):	0		
OVERALL REPLACEMENT FREQ. FACTOR:	1.00	SYSTEM IMPACTS INCLUDED(Y=1,N=0):	0		
SPECIFIC REPLACEMENT FREQ. FACTOR:	1.00	REBOOST COST CALCULATIONS			
POSTULATED LIFE CYCLE PROGRAM		ORBIT ALTITUDE(n.mi):	250		
TIME RATE FACTOR:	.00%	DENSITY SELECTION NUMBER:	2		
LIFE CYCLE TIME(yr):	30	IPS-SPECIFIC IMPULSE(lb-s/lb mass):	230		
MAXIMUN POWER GROWTH(KWe):	300	CDN-MAXIMUM DRAG COEFFICIENT:	2.30		
		LAUNCH COST FACTOR:	1.00		
		REBOOST FUEL PALLET FACTOR:	1.70		

FILE NO: DP460ML ROCKETDYNE
REVISION DATE: 18-Jun-86 R.PASCHALL X4796

OPERATION, MAINTENANCE, LOGISTICS WORKSHEET SUMMARY

COST ELEMENTS	MODULE NO. / TYPE / POWER(KWe)	STATION 1 PV 12.50	2 CBC 25.00	3 ORC 25.00	STATION 4 COM PMAD IOC	STATION 5 COM PMAD 250.00 DELTA GROW	PLATFORM 6 PV 8.00	PLATFORM 7 COM. PMAD NO GROWTH	8 NO GROWTH
RPLCMNT HARDWARE/CONSUMABLES(lb mass/yr)									
RPLCMNT HARDWARE/CONSUMABLES COST($M/yr)									
RPLCMNT HARDWARE/CONSUMABLES LAUNCH COST($M/yr)									
COST SUBTOTAL($M/yr)									
EVA TIME(h/yr)									
IVA TIME(h/yr)									
OPERATIONS AND MAINTENANCE COST($M/yr)									
DEPLOYMENT($M)									

ORBIT ALTITUDE (n.mi):	250.00
LAUNCH COST($/lb mass):	2920.00
LAUNCH PACKAGE MASS FACTOR:	1.20
EVA COST($K/h):	103.00
IVA COST($K/h):	15.20
*SUBCONT. & RK G&A FACTOR:	1.50
*MGMT.,SE&I,LOGISTICS,TEST AND GSE FACTOR:	1.50
LAUNCH COST FACTOR:	1.00
OVERALL RPLCMNT FREQ. FACTOR:	1.00
SPECIFIC RPLCMNT FREQ. FACTOR:	1.00

Table 8-4. Example of OML Worksheet Summary

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		MASS ASSESMENT SHEET							
		MODULE MASS (lb mass)							
		STATION			STATION	STATION	PLATFORM	PLATFORM	
EPS COMPONENT	MODULE NO.	1	2	3	4	5	6	7	8
	TYPE	PV	CBC	ORC	COM PMAD	COM PMAD	PV	COM. PMAD	NO GROWTH
	POWER(KWe)	12.50	25.00	25.00	IOC	250.00 DELTA GROW	8.00	NO GROWTH	
P6S-PV		1808.00	0.00	0.00	0.00	0.00	1808.00	0.00	
P6S-SD(CBC/ORC)									
CONCENTRATOR		0.00	1948.00	2181.00	0.00	0.00	0.00	0.00	
RECEIVER		0.00	3367.00	2624.00	0.00	0.00	0.00	0.00	
POWER CONVERSION UNIT		0.00	1527.00	652.00	0.00	0.00	0.00	0.00	
RADIATOR		0.00	1927.00	3457.00	0.00	0.00	0.00	0.00	
SUBTOTAL P6S-SD		0.00	8769.00	8914.00	0.00	0.00	0.00	0.00	
BETA JOINT ASSY		1300.00	650.00	650.00	0.00	0.00	0.00	0.00	
P6S TOTAL		3108.00	9419.00	9564.00	0.00	0.00	1808.00	0.00	
ESS TOTAL NI-H2 BATTERY		2792.00	0.00	0.00	0.00	0.00	1005.00	0.00	
PMAD-STATION PV SOURCE		559.00	0.00	0.00	0.00	0.00	0.00	0.00	
PMAD-STATION SD SOURCE		0.00	109.00	109.00	0.00	0.00	0.00	0.00	
PMAD-STATION COMMON		0.00	0.00	0.00	10299.20	9579.60	0.00	0.00	
PMAD-PLATFORM PV SOURCE		0.00	0.00	0.00	0.00	0.00	895.00	0.00	
PMAD-PLATFORM COMMON		0.00	0.00	0.00	0.00	0.00	0.00	1016.10	
PMAD TOTAL		559.00	109.00	109.00	10299.20	9579.60	895.00	1016.10	
TOTALS		6459.00	9528.00	9673.00	10299.20	9579.60	3708.00	1016.10	

Table 8-5. Example of Mass Assessment Worksheet Summary
(values not current)

9.0 RECOMMENDED CHANGES TO REQUIREMENTS

During the Phase B preliminary design of the Electrical Power System (EPS), Rocketdyne and NASA-LeRC have maintained close communication on all technical matters. Design requirements for the EPS have gone through several iterations as the preliminary design matured and Rocketdyne has continually reviewed each iteration and provided recommendations for changes as part of the on-going technical exchange.

The recommended changes to requirements documented in Rocketdyne's previous DR-02 submittal could be categorized into two types; (1) requirements dealing with the general subject of platform and station solar array sizing; and (2) specific design requirements. All of the recommended changes to specific design requirements have since been incorporated into the current requirements document and will not be repeated in this DR-02 submittal. Only the solar array sizing criteria and currently outstanding recommendations will be discussed.

9.1 SOLAR ARRAY SIZING CONSIDERATIONS

The following two items are repeated from the 30 June 1986 DR-02 submittal because of their continuing impact on the baseline design configuration.

9.1.1 Solar Array Sizing

The Power System Definition and Requirements Document specifies that the station solar array wing size be determined by the polar platform sizing requirements. This requirement results in a non-optimized station design, with each station PV power module capable of providing only 11.75 kwe net to the user interface under the required sizing criteria. The SD power modules must therefore be sized to provide 25.75 kwe net each to meet the total station requirement of 75 kwe.

While these sizes can be considered to meet the requirement of a nominal 25/50 power split, several other requirements are affected and must be considered:

1) Man-tended station would be 23.5 kwe, not 25 kwe as specified,

2) Failure tolerance for the station would be 75/49.25/23.5/0, not 75/50/25/0 as specified,

3) Power needs during station buildup and assembly should be carefully considered since less PV power would be available, and

4) Packaging of SD modules in the orbiter would become more difficult due to their larger size.

Therefore, sizing of the station PV output for the full 25 kwe, and accepting a slight mass penalty on the platform, is recommended for consideration.

9.1.2 Solar Array Commonality

This requirement, related to the discussion in section 9.1.1, specifies that identical solar arrays be used on the station and platform. The impacts of imposing this requirement are discussed in section 9.1.1. Although Rocketdyne recognizes and supports the importance of commonality, we recommend that consideration in this case be given to having arrays with identical cells, panels, mast assemblies, container assemblies etc., but having a different number of panels on the platform and station solar array wings. This would retain most of the cost benefits attributable to commonality, while allowing the optimum power output to be chosen independently for the platform and station.

9.2 EPS DESIGN REQUIREMENTS

The following recommended requirements changes are based on the latest marked-up requirements document received by Rocketdyne on 10 September 1986. Recommended changes to this document have been formally transmitted to NASA-LeRC in several letters and are summarized below.

9.2.1 Station Altitude (Paragraph 1.1.1B)

The current EPS design is based on a station altitude range of 180-250 nmi. Operation at any altitude outside of this range would require resizing of EPS components to meet performance requirements.

9.2.2 Peaking (Paragraph 1.2.1.2)

The peaking requirements, as currently stated, leave several questions unanswered which could impact the EPS design. It is recommended that additional clarification of these requirements include answers to the following questions.

- Can there be more than two peaks per orbit?
- Will there be prior knowledge of peaking, size of peak, duration of peak, etc.?
- Will there be knowledge of peak before start of peaking orbit?

9.2.3 Contingency (Paragraph 1.2.1.3A)

It is recommended that the period of time allowable before return to full power after a contingency period be stated. We suggest one orbit with an interim requirement of contingency-level power. This will allow time for restart of the SD modules and recharging of the batteries after the contingency is over.

9.2.4 PMAD Component Cooling (Paragraph 1.6.1C)

It is recommendeded that the PMAD component cooling requirements be clarified such that PV source PMAD which is attached to the battery cold plate in the PV equipment box have the same thermal limits as required by the battery.

9.2.5 Battery ORU Definition (Paragraph 2.2.2H)

It is recommended that the battery ORU definition be modified to state that each battery "assembly" constitutes an ORU. This would be consistent with Rocketdyne's definition which is that 23 cells constitute a battery assembly and four battery assemblies constitute a battery.

9.2.6 Concentrator Optical Requirements (Paragraph 2.3.1E)

It is recommended that the concentrator optical specular reflectance requirement be modified by the addition of "due to atomic oxygen degradation". This would remove from the requirement the possible effects of contamination beyond our control.

9.2.7 CBC Turbine Inlet Temperature (Paragraph 2.3.3.1C)

To reflect the current design, the turbine inlet temperature should be changed to a nominal operating range of 1370 - 1450^{o}F.

9.2.8 Rice - Lundell Alternator Outputs (Paragraphs 2.3.3.1D, 2.3.3.2D)

To reflect the current design, the Rice-Lundell alternator outputs for the CBC and ORC power conversion units should be changed to 208 VAC, 3 phase, 1067 Hz and

208 VAC, 3 phase, 1296 Hz respectively.

9.2.9 CBC Alternator Cooling (Paragraph 2.3.3.1E)

To reflect current design, the alternator cooling should be accomplished by the working fluid and liquid coolant.

9.2.10 ORC Requirements (Paragraph 2.3.3.2G)

For consistent terminology it is recommended that "recuperator" be changed to "regenerator".

9.2.11 PMAD Source Architecture (Paragraph 2.4.1C)

To clarify this requirement, it is recommended that it be changed to read "Frequency converters and inverters are capable of being operated either in a parallel or non-parallel mode".

9.2.12 PMAD Bus and PDCU Sizing (Paragraphs 2.4.2B, 2.4.3B, 2.4.5C)

It is recommended that the bus sizes and PDCU capacity specified in these paragraphs not be specified until the maximum demand loads are determined, and TBD be substituted at the present time.